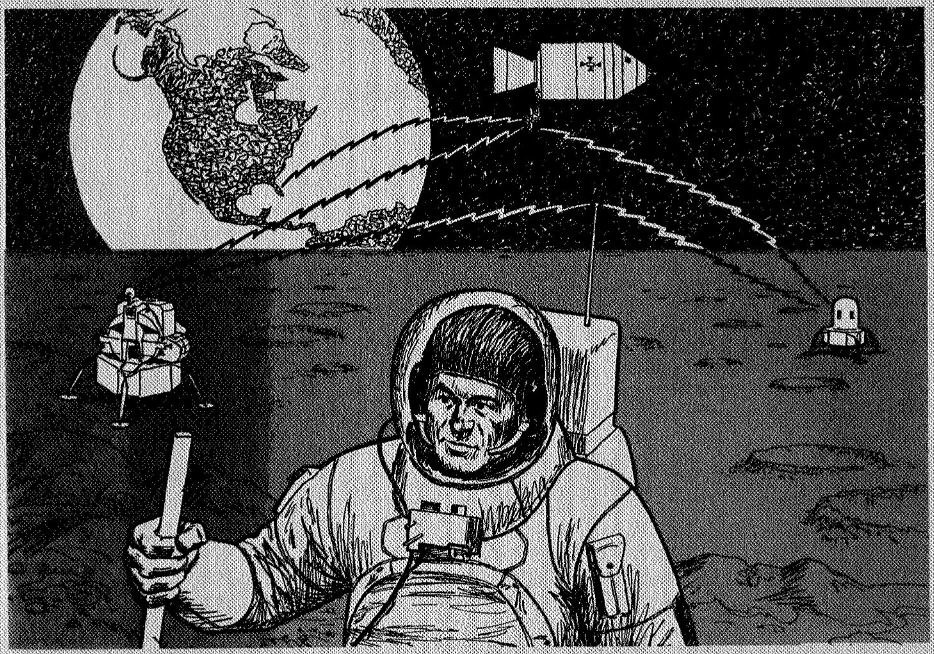
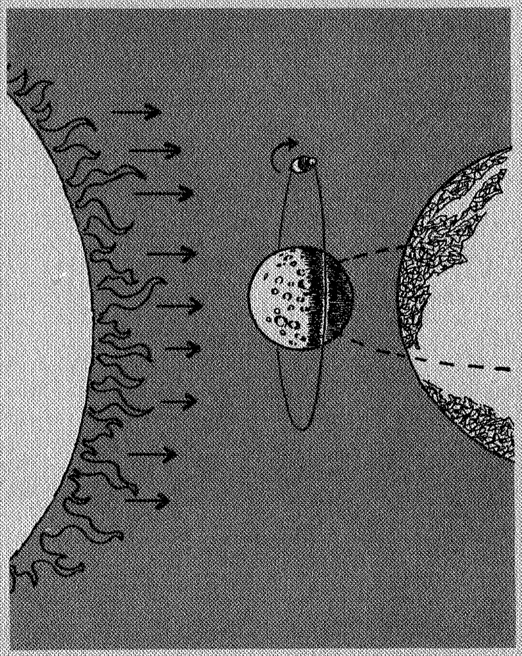
SD 68-850-2





Extended
Lunar Orbital Rendezvous Mission

VOLUME II - SUPPLEMENTAL DATA

SPACE DIVISION OF NORTH AMERICAN ROCKWELL CORPORATION

SD 68-850-2

A STUDY OF AN EXTENDED LUNAR ORBITAL RENDEZVOUS (ELOR) MISSION

CONTRACT NAS2-4942

FINAL REPORT

VOLUME II SUPPLEMENTAL DATA

January 1969

Prepared by

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SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

FOREWORD

This is Volume II of a three-volume report recording the results of a study of the application of data derived under a study of Space-Mission Duration-Extension Problems to an Extended Lunar Orbital Rendezvous Mission, hereafter referred to as ELOR. The three volumes of this report are:

Volume I	Technical	SD 68-850-1
Volume II	Supplemental Data	SD 68-850-2
Volume III	Summary of Results	SD 68-850-3

This volume presents some of the supplemental data generated throughout the study period, particularly by the subcontractors. It is presented because of its direct bearing on the study findings and serves to establish the validity of the results.

The study was conducted under Contract NAS2-4942 for the Mission Analysis Division of the Office of Advance Research and Technology (OART), National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California.

The work was performed under the direction of Roy B. Carpenter, Jr., Study Manager. Substantial contributions were made to this study by the following subcontractors and personnel thereof who provided the data at no cost, either for this study or the former baseline study.

A. C. Electronics	Al Lobinstine
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*Data supplied for baseline study

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The study was based on data derived from the baseline study, a company funded effort documented under NASA Contract NAS2-4214, and the mission systems design derived by the Lockheed Missiles and Space Company (LMSC) under Contract NAS8-21006.

*Data supplied for baseline study
**Study funded under NAS9-6608

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TABLES

ABBREVIATIONS AND ACRONYMS

The following list contains abbreviations and acronyms used in the Appendixes presented in this volume. These are presented in alphabetical order for easy reference.

A/D	Analog to digital
AGC	Automatic gain control
ALSEP	Apollo Lunar Surface Experiment Package
bps	Bits per second
C&D	Communications and data
CDD	Command demodulator-decoder
CDU	Coupling data unit
CM	Command module
CMC	Command module control
C/N	Carrier to noise ratio
CNR	Carrier to noise ratio
CSM	Command-service modules
CSS	Communications and status system
D/A	Digital to analog
DAC	Direct and alternating current
dbm	Decibels below 1 milliwatt
DRG	Digital ranging generator
DSIF	Deep Space Instrumentation Facility
ECS	Environmental control system
EKG	Electrocardiogram
ELOR	Extended Lunar Orbit Rendezvous
ELS	Earth landing system
EMCA	Electrical monitoring and control assembly
EMU	Extravehicular mobility unit
E/No.	Energy to noise power spectral density ratio
EPS	Electrical power system
EVC	Extravehicular communication system
FC	Fuel cell
FDAI	Flight director attitude indicator
FSK	Frequency shift keying

G&N	Guidance and navigation system
GLFC	Graphite lunar fuel capsule
HS	Heat shield
HX	Heat exchanger
Hz	Hertz (cycle per second)
IF	Intermediate frequency
IG	Inertial guidance
IGA	Inner gimbal angle
IMU	Inertial measurement unit
IRIG	Inertial rate integrating gyro
ISS	Inertial subsystem
LM	Lunar module
LMSC	Lockheed Missiles and Space Co.
MG	Manual guidance
MGA	Middle gimbal angle
MRC	Moisture recovery control
MSFN	Manned Space Flight Network
MTVC	Manual thrust vector control
OART	Office of Advanced Research and Technology
OG	Optical guidance
OGA	Outer gimbal angle
PA	Power amplifier
PAM	Pulse amplitude modulation
PIPA	Pulsed integrating pendulous accelerometer
PLSS	Portable life support system
PM	Pulse modulation
PMP	Premodulation processor
PRI	Primary
P_{sc}	Power in subcarrier
P_{total}	Total received power
RCC	Reactant conditioning control
RCS	Reaction control system
RMS	Root mean square
RTG	Radioisotope thermoelectric generator
SCS	Stabilization and control system
SEC	Secondary

SME	Status monitoring equipment
SNR	Signal to noise ratio
SPA	Signal processor assembly
SPS	Service propulsion system
TCC	Temperature conditioning control
TM	Thermal measurements
TVC	Thrust vector control
UAC	United Aircraft Corp.
UDL	Up-data link
USBE	Unified S-band equipment
VCO	Voice carrier output
VLV	Valve
VSWR	Voltage standing wave ratio
WRC	Water recovery control

APPENDIX A

RECOMMENDED CM-ECS OPERATIONAL PROCEDURES FOR THE ELOR MISSIONS*

REQUIREMENTS FOR THE CM-ECS STORAGE MODE

Prior to deactivation of the suit circuit, most of the waste water should be removed from the suit heat exchanger. This is accomplished by cycling one cyclic accumulator and then setting the water service valve on the heat exchanger to the SEP position. In the SEP position, the water service valve prevents loss of water from the cyclic accumulator and associated water inlet line, so that the retained water could be used for subsequent priming of the suit heat exchanger when the suit circuit is reactivated.

To facilitate the priming of the exchanger upon reactivation, a small (2 to 3 pound) water tank has been added to the ECS. A manual valve (Item 5. 18) connected between the water tank and the potable water supply circuit is used to fill and then isolate the water tank. A second valve at the tank outlet is used to flow water to the water service valve on the suit heat exchanger. The water service valve is kept in the SEP position, and the water valves are kept in the closed position during the dormant orbital mode.

The suit-circuit return check valve (Item 1. 3) is maintained in the "0" or open position and the suit hose connector assemblies (Item 1. 34) and demand pressure regulator (4. 16) are in the off position. The open suit circuit check valve allows the suit circuit to be maintained above the cabin pressure, but will not allow the reverse situation. When the crew departs from the CM, the internal pressure is 5 psia. No attempt is made to lower this pressure directly, since this would be wasteful of oxygen. Instead, the pressure level is allowed to leak down to the 0. 5 psia level. Similarly, the suit circuit is maintained for some time at 5 psia before leakage brings it down to the 0. 5 psia level. Due to this low rate of leakage, the suit circuit pressure lags the cabin pressure. In the event of cabin decompression, suit circuit pressure should be maintained for considerable time. Upon cabin repressurization, the open check valve assembly allows immediate repressurization of the suit loop. Of course, both compressors are off.

Both CO_2 absorber elements are removed from the absorber canister to prevent the elements from swelling due to the absorption of residue water evaporated from the suit heat exchanger.

*Provided by AiResearch Corp. through Reference 4. 4 of Vol. I.

The oxygen flow valve (Item 4. 22) is kept in the normal flow position. The solenoid shutoff valve is closed to isolate the demand pressure regulator (Item 4. 16), the cabin pressure regulator (Item 3. 28), and the emergency inflow valve (Item 4. 22) during the dormant orbital mode. Since the inflow valve is open when the cabin pressure is below 4. 6 psia, the cabin can be pressurized from 0. 5 to 5. 0 psia by remotely opening the oxygen solenoid shutoff valve to transfer between the CM and the LM.

The glycol reservoir remains isolated by the glycol shutoff valves in the glycol reservoir flow loop. All unused controls are turned off, while the pump selector control is activated.

ECS DEACTIVATION PROCEDURE

1. Activate one cyclic accumulator five times at approximately 4-minute intervals to expel most of the water from the suit heat exchanger. Then set the selector switch for the cyclic accumulator system to OFF. Make sure that the downstream and upstream shutoff valves for the suit-circuit-exchanger water tank are closed; then set the water service valve on the suit heat exchanger to the SEP position.

2. Energize the ECS/EPS coolant interface solenoid valves (refer to Figure 3-1) to connect the ECS and EPS coolant circuits.

3. Turn off the electrical power to the glycol temperature control (Item 2. 22) and then set the glycol temperature control valve (Item 2. 42) to CLOSE position.

4. Set the glycol manual diverter valve (Item 2. 36) in the primary and secondary coolant circuits to the radiator bypass position. Prepare the ECS space radiator control system for the dormant orbital mode.

5. Deactivate the cabin temperature control (Item 3. 7) and set the cabin temperature control valve (Item 2. 13) to the heat position.

6. Activate the automatic selector control equipment for the glycol pump assembly. Operate the remote control equipment to check the transfer of power from the operating pumping unit to the redundant pumping unit.

7. Activate the cabin low-pressure transducer.

8. Adjust the glycol manual metering valve (Item 2. 35) in the secondary coolant circuit to the ON position for full flow to the cabin heat exchanger.

9. Turn off the power to the suit compressor (Item 1. 10).

10. Disconnect suit hoses from the suit hose connector assemblies (Item 1. 34) and set the flow control levers on the latter to OFF position.

11. Set the suit test and inlet selectors on the demand pressure regulator (Item 4. 16) to OFF position.

12. Place the primary gylcol evaporator temperature control system in the nonoperating mode.

13. Remove the two CO_2 absorber elements from their canister and place the elements in the CO_2 absorber stowage compartment.

14. Leave the suit circuit return check valve (Item 1. 3) in the "0" or open position and the emergency inflow valve, cabin pressure regulator (Item 3. 28), and demand pressure regulator from the cabin gas circuit. Open the manual valve to the cabin pressurization restrictors.

ESC REACTIVATION PROCEDURE

1. Pressurize the CM by either remotely activiating the oxygen shutoff valve to OPEN position, or by opening a valve on the forward hatch to pressurize the CM with oxygen from the LM. When the oxygen shutoff valve is opened, oxygen is supplied to the emergency inflow valve (Item 4. 22) and the cabin pressure regulator (Item 3. 28) for release to the cabin. Close the shutoff valve to the cabin pressurization restrictor.

2. After entry into the CM, place the primary glycol evaporator temperature control system in the automatic mode.

3. Install the CO_2 absorber elements into the absorber canisters. Set flow control levers on suit hose connector assembly to cabin position.

4. Set the inlet selector on the demand pressure regulator to the 1 and 2 position.

5. Activate one suit compressor.

6. Open the shutoff valve downstream of the new added water tank to allow the stored water to flow through the water service valve to the suit heat exchanger. Then close the shutoff valve and set the water service valve to the OFF position.

7. Set the selector switch for either one of the cylic accumulator systems to the auto position.

8. Connect the suit hoses to the suit hose connector assembly.

9. Deactivate all remote controls and unnecessary instrumentation.

10. Activate the ECS space radiator system.

11. Activate the glycol temperature control (Item 2. 22).

12. Set the glycol manual diverter valves (Item 2. 36) in the primary and secondary coolant circuits to the normal position to allow coolant to flow to the ECS space radiator.

13. Deenergize the ECS/EPS coolant interface solenoid valves to isolate the ECS coolant circuits from the EPS coolant circuit.

14. Operate the cyclic accumulator as required to expell excess water from the suit heat exchanger.

APPENDIX B

INERTIAL INSTRUMENT TEMPERATURE CONSIDERATIONS, P/O G&N SYSTEM, CSM AND LM*

Under present Apollo operational procedures, the IMU heater power is always applied whether the system is operational or not. Long storage periods for both the CM and LM systems in the standby mode could represent serious reliability and power drain problems. However, considerable evidence exists that inertial instrument performance degradation does not occur after cooldown of the instruments from their operating temperatures. This would enhance the feasibility of using the Apollo G&N system for the proposed mission, the only question being, what temperature limits do exist for the inertial instruments and the G&N equipment as a whole without appreciable performance degradation.

The gyros were designed to allow storage at room ambient conditions. During buildup and after final assembly, the temperatures of the 25 IRIG's and the 16 PIPA's both are allowed to cool to room ambient (70 F) numerous times in the normal sequence of events. According to the procurement specification (Reference 4) applying to the purchase of these instruments from NASA, the storage temperature of the gyros shall be 70 ±10 F and the accelerometers shall be between 120 F and 142 F.

Once installed in an IMU, temperatures of 135 ±2.5 F for the gyros and 130.5 ±1.5 F for the accelerometers are to be maintained. In the event of an IMU cooldown, the procedures of MIT/IL ISS Memorandum 400 (Reference 5) are to be followed. These state that any gyro temperature outside the range of 20 F to 160 F and any PIPA temperature outside the range of 120 F to 140 F requires recalibration. The memorandum further states that gyro temperatures outside the range -40 F to 170 F and PIPA temperatures outside the range 40 F to 160 F may cause permanent damage.

In practice, it has been found that the listed limits for recalibration are somewhat arbitrary because they have at times been exceeded inadvertently or under special tests with no apparent change in calibration or degradation in instrument performance. Also, 25-IRIG gyros have been cooled down to -65 F without any permanent damage (Reference 4.7.)

*Provided by the AC Div. of GM through Reference 4.7 of Vol. I.

Cooldown of the IMU during non-operational periods to room ambient conditions or as low as 40 F appears feasible since no permanent damage should occur, and the gyros, at least, would not require calibration. The sensitivity of accelerometer coefficients to temperature cycling must be investigated further to determine if the resulting deviations can be tolerated or if recalibration would be required in the lunar area.

Accelerometer bias could be determined on the moon by measuring the accelerometer null outputs and in orbit by measuring the accelerometer output under coasting conditions. Any bias changes can then be compensated for in the respective LM and CM computers. Scale factor determination may be performed on the lunar surface as on the earth, however, it is impractical for the CM system in lunar orbit.

If storage temperatures much lower than room ambient are planned, further investigation on the PIPA's as well as other G&N equipment would be desirable to determine exactly what the limiting factors are.

APPENDIX C

CSM S-BAND OMNI-DIRECTIONAL ANTENNAS FOR THE EXTENDED LUNAR ORBIT MISSION*

This Appendix evaluates the adequacy of the present Apollo CSM S-band omnidirectional antennas for utilization on the extended lunar orbit mission (ELOM).

First, it is desirable to evaluate the need for an additional S-band omnidirectional antenna. It recommended that an additional S-band omnidirectional antenna be provided principally for lunar surface to command module communications. However, S-band communications capability does not presently exist between the CSM and LM (i.e., the CSM receives at 2106 MHz and transmits at 2287.5 MHz while the LM transmits at 2282.5 MHz and receives at 2101 MHz). Therefore, an additional S-band omnidirectional antenna is not a solution because of incompatibility between the LM and CSM communication subsystems.

The primary antenna problem that must be considered is the look angle at the CSM antennas while in unattended lunar orbit from the earth-based MSFN** stations. Deep nulls in the antenna pattern will be observed if the longitudinal axis of the CSM points at or near the MSFN station. Thus, the orientation of the CSM with respect to the MSFN ground stations must be considered to determine if the present S-band omnidirectional antennas are adequate. The S-band high-gain antenna is not considered due to the spin rate of the CSM which would severely limit communications between the CSM and MSFN.

Considering the MSFN station located furthest from the equator in a north-south direction would present the worst case ground station location for pointing considerations. The Madrid station located at 40.4 north latitude is the MSFN station furthest from the equator when considering the stations with an 85-foot diameter antenna capability which is required for the ELOR. This station can be separated by as much as 63.7 degrees from the ecliptic plane. In order for the Madrid MSFN antenna to point its boresight axis parallel to the ecliptic plane, the antenna must be depressed 63.7 degrees from its zenith. The orientation of the moon varies approximately plus or minus 5.2 degrees with respect to the ecliptic plane. The CSM's longitudinal axis will

*Provided by the Collins Radio Co. through Reference 4.10 of Vol. I.
**Manned Space Flight Network

always be nearly normal to the ecliptic plane since the distance between the lunar orbit of the CSM and the sun is very large. Based upon this worst case geometry, the Madrid station would have to depress its antenna 68.9 degrees below the zenith to illuminate the moon and the spacecraft. The angle between the CSM's longitudinal axis and the MSFN antenna boresight would be 84.8 degrees. All other earth-moon geometries will present an angle within 90 degrees plus or minus 5.2 degrees. Considering this geometry along with the lunar polar orbit and stabilization of the CSM with respect to the sun assures that the longitudinal axis will never be pointed at the MSFN stations. This broadside look angle at the CSM will present the most favorable antenna patterns. Thus, it appears that the existing S-band omnidirectional antennas are sufficient for the ELOR communications functions when operated at lower data rates.

APPENDIX D

PHASE-LOCK RECEIVER ACQUISITION DURING CSM UNATTENDED LUNAR ORBIT*

The unattended lunar orbit phase of the ELOR will require new acquisition procedures to lock the spacecraft's USBE phase-locked receiver to the MSFN carrier.

Presently the MSFN first locks the up-link by sweeping the ground transmitter over a frequency range sufficient to lock the spacecraft receiver. Upon lockup of the spacecraft receiver the spacecraft transmitter will be following the sweep generated by the ground transmitter. The down-link carrier continues to follow the up-link sweep until the ground receiver has locked up. Two-way lock is recognized by the ground station operator noting that the static phase error of the ground receiver is following the transmitter sweep. After lockup of the ground receiver, the MSFN sweep automatically decays to permit transmission at the nominal transmit frequency. Approximately eight seconds is required to attain two-way RF acquisition.

It is assumed, for the purposes of this study that at least one MSFN dual station will be solely dedicated to the ELOR during all periods of the mission (i. e., a ground network is available for a continuous communication capability). To meet the combined status monitoring and remote control function requirements, it appears desirable to have the MSFN station within line of sight of the moon provide a continuous beacon carrier signal for the purpose of antenna selection.

Since the CSM will be spinning at a rate of 0.5 revolutions per minute, the USBE's two phase-lock receivers after initially acquiring, may be alternately locking up and unlocking. The CSM S-band omnidirectional antenna patterns, modified as recommended in Section 4.1.5 of Volume I, are not well enough defined to make an accurate estimate as to the ability of a single receiver to maintain lock during a complete revolution. The nominal received S-band uplink signal level is approximately minus 100 dbm assuming that an 85-foot diameter antenna is used at the MSFN ground station. The USBE receiver will remain locked to signals levels of minus 130 dbm. Thus a 30-db degradation in antenna gain due to a poor orientation of the CSM with respect to the ground station will still provide a sufficient signal level for the receiver to remain locked. If the antenna gain was degraded more than 30 db, the receiver possibly may lose lock. Depending on the period of time that the antenna gain is degraded more than 30 db, a

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possibility exists that the spacecraft's receiver's frequency may not have drifted far from the ground transmitter and reacquisition without sweeping the uplink carrier may be possible. The foregoing considerations require detailed study to determine an acceptable acquisition procedure.

One acquisition procedure that can be considered is a method that would employ a continuous sweep of the uplink beacon signal except during command transmissions. This would ensure that at least one of the spacecraft receivers would be in lock or could acquire within one sweep of the carrier. (This is based upon the USBE specification requirement for a 90-percent probability of acquisition when the received signal is swept linearly at a rate of minus 114 dbm.) When it is desired to transmit a command to the spacecraft, the sweep would be decayed to the nominal carrier frequency and followed by modulation of the carrier by the up-data subcarrier. The first command would be to turn the downlink S-band PM transmitter ON. The ground station would then start sweeping the carrier again to enable the ground receiver to lock-up. After the ground receiver had locked, the sweep would again be decayed and followed by modulation of the carrier by the up-data subcarrier. The second command would be to activate the SME data link for command verification. The third and all following up-data messages could be actual commands to the spacecraft systems as processed by the up-data link. The command verification rate will be limited by the capacity of the SME data link.

To summarize this acquisition procedure, Table D-1 lists each operational step along with a time estimate for each step. Thus, the first true command would require a time duration of nominally 25 seconds after the initiation of the first command at the ground station (17.3 seconds for acquisition, 2.6 seconds for two-way propagation delay and nominally 5 seconds for verification). Each succeeding command will require approximately 12.6 seconds. If verification is not required, commands may be sent at the same rate as presently used for the Apollo mission (i.e., 200 information bits per second).

Table D-1. ELOR S-Band Acquisition Procedure for Up-Link Commands

Operation	Time (seconds)
One-way lock	6.5
Command Transmitter ON	2.5
Two-way lock	4.6*
Activate SME data link	2.6
Total time for acquisition	17.3
Each command is limited by SME data link capacity to one command per ten seconds** plus two-way propagation time delay.	

*Assumes MSFN can lock within 3 seconds after receipt of swept downlink carrier.

**Assumes one SME data channel (S consecutive bits) available for command verification per data frame.

APPENDIX E

CIRCUIT QUALITY EVALUATION, CSM TO MSFN DATA LINK*

The expected power calculations are presented in Table E-1. The following data were used in the analyses:

Transmitter power output of 275 mw.

Transmitter circuit loss was calculated as follows:

Cable and connector loss**	5.2 db
Cable VSWR loss**	0.3 db
Diplexer-power divider loss	4.3 db
Antenna switch**	0 3 db
RF switches within USBE	1.0 db
Total =	11.1 db

Assumed antenna gain.

For a bit error probability of 10^{-3}, an SNR of 7.4 db is required in a bandwidth equal to the bit rate (40 bps). The 7.4 db requirement has been degraded by 1 db over the theoretical value given in Figure 10 of Reference E-1 for practical circuit implementation considerations.

A 1.6 radian peak phase deviation will maximize the portion of total power available for the information sidebands and still allow an adequate portion of the total power to remain in the carrier component to assure lock. The modulation loss may then be computed as follows since the first order sidebands of the modulated carrier will add coherently. (See Reference E-2.)

$$\frac{P_{sc}}{P_{total}} = 2\left[J_1 (\theta) \right]^2 = 2\left[0.57 \right]^2 = 0.644$$

$$P_{sc} = 0.644 \; P_{total}$$

Modulation loss log10 0.644 = -1.93 db

*Provided by the Collins Radio Co. through Reference 4.10 of Vol. I.

**From NR, Space Division, Interface Control Document (ICD) MH01-13001-414 (A)

Table E-1. CSM to MSFN SME Data Link

Parameter	Nominal Value	Notes
Computation of received carrier power		
Transmitter carrier power (dbm)	-5.6	(1)
Transmitter circuit losses (db)	11.1	(2)
Transmitting antenna gain (db)	0	(3)
Polarization loss (db)	0.1	(4)
Free space loss (db)	211.4	(4)
Receiver antenna gain (db)	52.0	(4)
Receiver circuit loss (db)	0.8	(4)
Received carrier power (dbm)	-177.0	
Computation of desired carrier power		
Equivalent receiver noise density (dbm/cps)	-204.3	(4)
Receiver noise bandwidth (db)	16.0	(5)
Equivalent receiver noise power (dbm)	-188.3	
Desired rms signal-to-rms noise ratio	7.4	(5)
Modulation characteristic (db)	-1.9	
Desired carrier power (dbm)	-179.0	
Nominal Margin (db) +2.0*		
*Worst-case tolerances are estimated to be approximately -3.5 db.		

REFERENCES

E-1. Glenn, A. B., I.R.E. Trans on Communication Systems, June 1960, "Comparison of PSK vs. FSK and PSK-AM vs. FSK-AM."

E-2. Martin, B.D., The Pioneer IV Lunar Probe: A Minimum Power FM/PM System Design, JPL Technical Report 32-215 (15 March 1962).

APPENDIX F

CIRCUIT-QUALITY ESTIMATES FOR THE CSM TO LUNAR SURFACE DATA LINK*

The circuit quality for the proposed CSM to lunar surface data link has been evaluated, and the results are presented in Table F-1. The following data were used in the analysis:

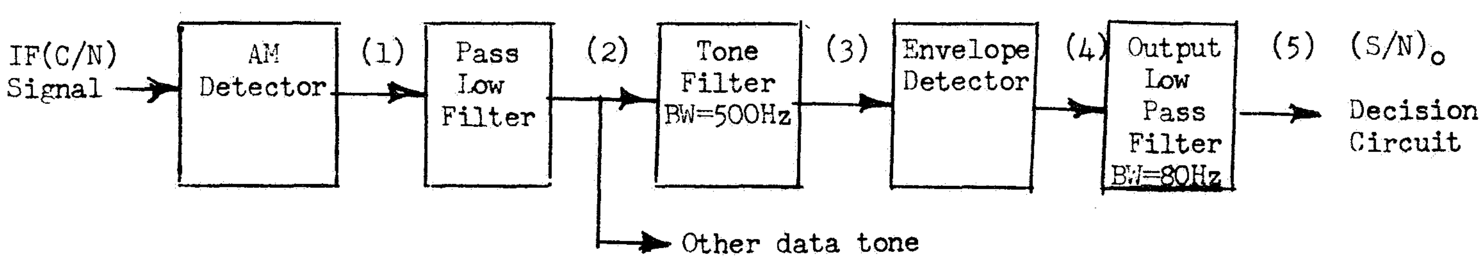
1. The CSM VHF/AM transmitter will provide a peak carrier power output of 10 watts if the internal noise suppression oscillator is disabled.

2. Derived from NR document "MH01-05113-414 (change C), CSM/LM VHF System Circuit Margin Calculations."

3. Assumed antenna gains.

4. Free-space loss for an RF carrier of 296.8 MHz at a range of 320 nautical miles (the slant range from the lunar surface to the CSM in an 80-nautical-mile lunar orbit will be 320 nautical miles at an elevation angle of 5 degrees above the lunar surface).

5. The modulator characteristic converts the IF carrier-to-noise ratio to which we then compare with the required signal-to-noise ratio. The modulation characteristic is calculated from an examination of the demodulator used to process the received signal.



LM VHF/AM Transmitter/Receiver, FSK Tone Demodulator

*Provided by the Collins Radio Co. through Reference 4.10 of Vol. I.

Table F-1. Circuit Quality Chart for CSM to Lunar Surface/LM SME Data Link

Parameter	Nominal Value	Notes
Computation of Received Carrier Power		
Transmitter carrier power (dbw)	10.0	(1)
Transmitter circuit losses (dbw)	5.2	(2)
Transmitting antenna gain (db)	-3.0	(3)
Polarization loss (db)	3.0	(2)
Free space loss (db)	137.8	(4)
Receiver Antenna gain (db)	0	(3)
Receiver circuit loss (db)	-3.2	(2)
Received carrier power (dbw)	-142.2	
Computation of Circuit Quality		
Equivalent Receiver noise density (dbw/cps)	-197.2	(2)
Receiver noise bandwidth (db)	48.5	(2)
Equivalent receiver noise power (dbw)	-148.7	
Received Carrier to Noise Power Ratio	6.5	
Modulation Characteristic (db)	12.7	(5)
Output Signal-to-Noise Power Ratio	19.2	
Desired Signal-to-Noise Power Ratio	16.2	(6)
Nominal Margin (db) 3.0*		

*Worst case tolerances are estimated to be -2.5 db.

The modulation characteristic for the CSM to lunar surface/LM SME data link is calculated as follows:

At point 1, the input C/N will be degraded by approximately 0.5 db due to the square law demodulators non-linear performance for low input carrier to noise ratios (CNR). (See Reference F-1.)

At point 2, the CNR will be improved by the ratio of the IF to twice the low pass filter bandwidth:

$$\frac{\text{IF Bandwidth}}{2 \times \text{Low-Pass-Filter Bandwidth}} = \frac{70,000 \text{ Hz}}{2(20,000)\text{Hz}} = 1.75 = +2.4 \text{ db}$$

At point 3, the CNR will further be improved by the ratio of the low-pass-filter bandwidth to the tone filter bandwidth.

$$\frac{\text{Low Pass Filter BW}}{\text{Tone Filter BW}} = \frac{20,000 \text{ Hz}}{500 \text{ Hz}} = 40 = +16.0 \text{ db}$$

At point 4, no degradation will be observed, since the CNR into the envelope detector is high.

At point 5, the output low pass filter will improve the CNR by the ratio of the tone filter bandwidth to twice the output low pass filter bandwidth.

$$\frac{\text{Tone Filter BW}}{2 \times \text{Output Low Pass Filter BW}} = \frac{500 \text{ Hz}}{2(80) \text{ Hz}} = 3.1 = 4.9 \text{ db}$$

The output signal-to-noise ratio must further be degraded since only a portion of the unmodulated carrier peak envelope power of 10 watts is available for the information sidebands which will pass through the tone filter. The combination of the FSK tone modulation and the ON-OFF AM characteristics of the VHF/AM transmitter will result in the useful sideband power being 10.1 db from the 10-watt peak envelope carrier output level. This value is obtained by considering the VHF/AM transmitter ON-OFF modulated by the FSK tone which is keyed ON for one half of the bit period. ON-OFF modulation by the square wave tones place half of the average power in the sidebands. The first-order tone harmonics will be the only information components which will pass through the tone filter resulting in 1.1 db. The resulting modulation characteristic is expressed below:

$$\text{Modulation Characteristic} = 0.5 \text{ db} + 2.4 \text{ db}$$

$$= 16 \text{ db} + 4.9 \text{ db} - 10.1 \text{ db} = +12.7 \text{ db}$$

6. Since the assumed output low-pass-filter bandwidth is equal to the reciprocal of the message pulse width, the signal-to-noise (SNR) is equal to the energy to noise power spectral density ratio (E/No). We assume that a bit-error probability of 10^{-5} will be sufficient for the mission requirements. This bit-error probability will result in the received message error rate of slightly more than one per 100,000 messages transmitted. The SNR (or E/No) required for the detection of an FSK signal by means of a non-coherent FSK demodulator at a 10^{-5} bit-error probability is 13.2 db per reference (2). This theoretical value is based upon utilization of a matched filter detection technique. We estimate that practical circuit considerations will degrade this value by approximately 3 db, resulting in a required SNR of 16.2 db.

REFERENCES

F-1. Fubini and Johnson, Proc. IRE December 1948, "Signal-to-Noise in AM Receivers"

F-2. Glenn, A. B., IRE Trans. on Communications Systems, June 1960, "Comparison of PSK vs. FSK and PSK-AM vs FSK-AN Binary Coded Transmission Systems"

APPENDIX G

CIRCUIT QUALITY ESTIMATES, MSFN TO CSM REMOTE CONTROL LINK*

The potential circuit quality for an MSFN to CSM remote control link has been evaluated and the results are presented in Table G-1. The following data were used in the analysis:

MSFN transmitter power output of 10 kw.

From NR-Space Division ICD MH01-13001-414(A)

Assumed antenna gain.

CSM receiver circuit loss is calculated as follows:

Cable and connector loss	5.2 db
Cable loss due to VSWR	0.3 db
Antenna-switch loss	0.3 db
Diplexer-power divider	4.3 db
RF switches within USBE	0.5 db
Total	10.6 db

*Provided by the Collins Radio Co. through Reference 4.10 of Vol. I.

Table G-1. MSFN to CSM Remote Control Link

Parameter	Nominal Value	Notes
Computation of received carrier power		
Transmitter carrier power (dbw)	40.0	(1)
Transmitter circuit losses (db)	0.5	(2)
Transmitting antenna gain (db)	51.0	(2)
Polarization loss (db)	0.1	(2)
Free space loss (db)	210.7	(2)
Receiver antenna gain (db)	0	(3)
Receiver circuit loss (db)	10.6	(4)
Received carrier power (dbw)	-130.9	
Computation of desired carrier power		
Equivalent receiver noise density (dbw/cps)	-190.5	(2)
Receiver noise bandwidth (db)	43.0	(2)
Equivalent receiver noise power (dbw)	-147.5	
Desired rms signal-to-rms noise ratio	10.0	(2)
Modulation characteristic (db)	1.7	(2)
Desired carrier power (dbw)	-145.8	
Nominal margin (db) 4.9*		

*Worst case tolerances are estimated to be 4.0 db.

APPENDIX H

CIRCUIT QUALITY ESTIMATE, LUNAR SURFACE TO CSM, REMOTE CONTROL LINK*

The circuit quality for the proposed lunar surface to orbiting CSM, Remote Control Link has been evaluated and the results are presented in Table H-1. The following data were used in the calculations:

1. The CSM VHF/AM transmitter will provide a peak carrier power output of 10 watts if the internal noise suppression oscillator is disabled.

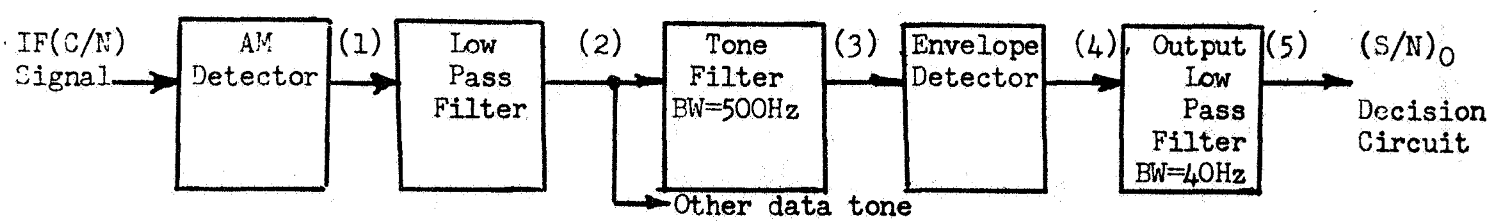
2. Derived from NR document "MH01-05113-414 (change C), CSM/LM VHF System Circuit Margin Calculations."

3. Assumed antenna gains.

4. Free-space loss for an RF carrier of 259.7 MHz at a range of 320 nautical miles. (The slant range from the lunar surface to the CSM in an 80 nautical mile lunar orbit will be 320 nautical miles at an elevation angle of 5 degrees above the lunar surface.)

5. The modulator characteristic converts the IF carrier-to-noise ratio to a usable output signal-to-noise to which we then compare with the required signal-to-noise ratio. The modulation characteristic is calculated from an examination of the demodulator used to process the received signal.

The assumed LM demodulator is shown below for one channel of the FSK data tones.



CSM VHF/AM Transmitter/Receiver, Command Demodulator-Decoder

*Provided by Collins Radio Company through Reference 4.10 of Vol. I.

Table H-1. Circuit Quality Chart for Lunar Surface-to-CSM Remote Control Link

Parameter	Nominal Value	Notes
Computation of Received Carrier Power		
Transmitter carrier power (dbw)	10.0	(1)
Transmitter circuit losses (db)	3.2	(2)
Transmitting antenna gain (db)	0	(3)
Polarization loss (db)	-3.0	(2)
Free space loss (db)	136.6	(4)
Receiver antenna gain (db)	-3.0	(3)
Receiver circuit loss (db)	4.2	(2)
Received carrier power (dbw)	-139.0	
Computation of Circuit Quality		
Equivalent receiver noise density (dbw/cps)	-198.0	(2)
Receiver noise bandwidth (db)	48.5	(2)
Equivalent receiver noise power (dbw)	-149.5	
Carrier to noise ratio	10.5	(5)
Modulation characteristic (db)	15.2	
Output signal to noise power ratio	25.7	
Desired signal-to-noise power ratio	16.2	(6)
Nominal Margin (db) 9.5*		
*Worse case tolerances are estimated to be -2.2 db		

The modulation characteristic for the CSM to lunar surface/LM SME data link is calculated as follows:

1. At point (1) the input C/N will be degraded by approximately 0.1 db due to the square law demodulators non-linear performance for low input carrier to noise ratios (CNR). (See Reference H-1.)

2. At point (2) the CNR will be improved by the ratio of the IF to twice the low pass filter bandwidth:

$$\frac{\text{IF Bandwidth}}{2 \times \text{Low Pass Filter Bandwidth}} = \frac{70{,}000 \text{ Hz}}{2(20{,}000) \text{ Hz}} = 1.75 = +2.4 \text{ db}$$

3. At point (3), the CNR will further be improved by the ratio of the low pass filter bandwidth to the tone filter bandwidth.

$$\frac{\text{Low Pass Filter BW}}{\text{Tone Filter BW}} = \frac{20{,}000 \text{ Hz}}{500 \text{ Hz}} = 40 = +16.0 \text{ db}$$

4. At point (4) no degradation will be observed since the CNR into the envelope detector is high.

5. At point (5) the low pass filter will improve the CNR by the ratio of the tone filter bandwidth to twice the output low pass filter bandwidth.

$$\frac{\text{Tone Filter BW}}{2 \times \text{Output Low Pass Filter BW}} = \frac{500 \text{ Hz}}{2(40) \text{ Hz}} = 6.0 = 7.0 \text{ db}$$

The output signal-to-noise ratio must be degraded since only a portion of the unmodulated carrier peak envelope power of 10 watts is available for the information sidebands which will pass through the tone filter. The combination of the FSK tone modulation and the ON-OFF AM characteristics of the VHF/AM transmitter will result in the useful sideband power being 10.1 db down from the 10 watt peak envelope carrier power output level. This value is obtained by considering the VHF/AM transmitter ON-OFF modulated by the FSK tone which is keyed ON for one half of the bit period. This results in an average transmitter output power of 2.5 watts. ON-OFF modulation by the square wave tone place half of the average power in the sidebands. The first order tone harmonics will be the only information components which will pass through the tone filter resulting in 1.1 db.

The resulting modulation characteristic is expressed below:

$$\text{Modulation Characteristic} = 0.5 \text{ db} + 2.4 \text{ db} + 16 \text{ db} + 7.8 \text{ db}$$

$$- 10.1 \text{ db} = +15.2 \text{ db}$$

6. Since the assumed output low pass filter bandwidth is equal to the reciprocal of the message pulse width, the signal-to-noise (SNR)

is equal to the energy to noise power spectral density ratio (E/No). We assume that a bit error probability of 10^{-5} will be sufficient for the mission requirements. This bit error probability will result in the received message error rate of slightly more than five per 1,000 messages transmitted. The SNR (or E/No) required for the detection of an FSK signal by means of a non-coherent FSK demodulator at a 10^{-5} bit error probability is 13.2 db per Reference H-2. This theoretical value is based upon utilization of a matched filter detection technique. We estimate that practical circuit considerations will degrade this value by approximately 3 db resulting in a required SNR of 16.2 db.

REFERENCES

H-1. Fubini and Johnson, Proc. IRE Dec 1948 "Signal-to-Noise in AM Receivers"

H-2. Glenn, A. B., IRE Trans. on Communications Systems June 1960, "Comparison of PSK vs. FSK and PSK-AM vs. FSK-AM Binary Coded Transmission Systems

APPENDIX I

CM COMMUNICATIONS AND DATA SYSTEM MODIFICATIONS AND NEW SYSTEMS DESCRIPTION, ELOR MISSION*

VHF/AM TRANSMITTER-RECEIVER

This must be modified to provide the receiver output to the Command Demodulator-Decoder and possibly a modulation input to accept the FSK data tones from the signal measurement equipment (SME).

The Apollo Block II unit presently accommodates the digital ranging signal on a modulation input channel that has sufficient bandwidth to handle the CSM to lunar surface/LM FSK tone signals. If the modulation input presently used for the digital ranging generator (DRG) is replaced with the SME data signal, no modification would be necessary for this change. If the DRG input is desired to be retained then time-sharing of the channel between the ranging and remote control function appears feasible since we cannot envision a requirement for simultaneous command and ranging. This modification would require an additional coaxial connector on the VHF/AM Transmitter-Receiver and solid state switching circuitry.

The VHF/AM Transmitter-Receiver must provide an output coaxial connector to route the receiver output to the Command Demodulator-Decoder.

The minimal VHF/AM Transmitter-Receiver changes required for ELOR are expected to have a negligible impact on weight or reliability of the present unit.

UNIFIED S-BAND EQUIPMENT

This must be modified as reflected in Figure I-1, and described in the following:

1. Simultaneous operation of both USBE receivers.
2. Independent ON-OFF controls for receiver and PM exciter of each transponder.

*Much of the data used herein was provided by the Collins Radio Company.

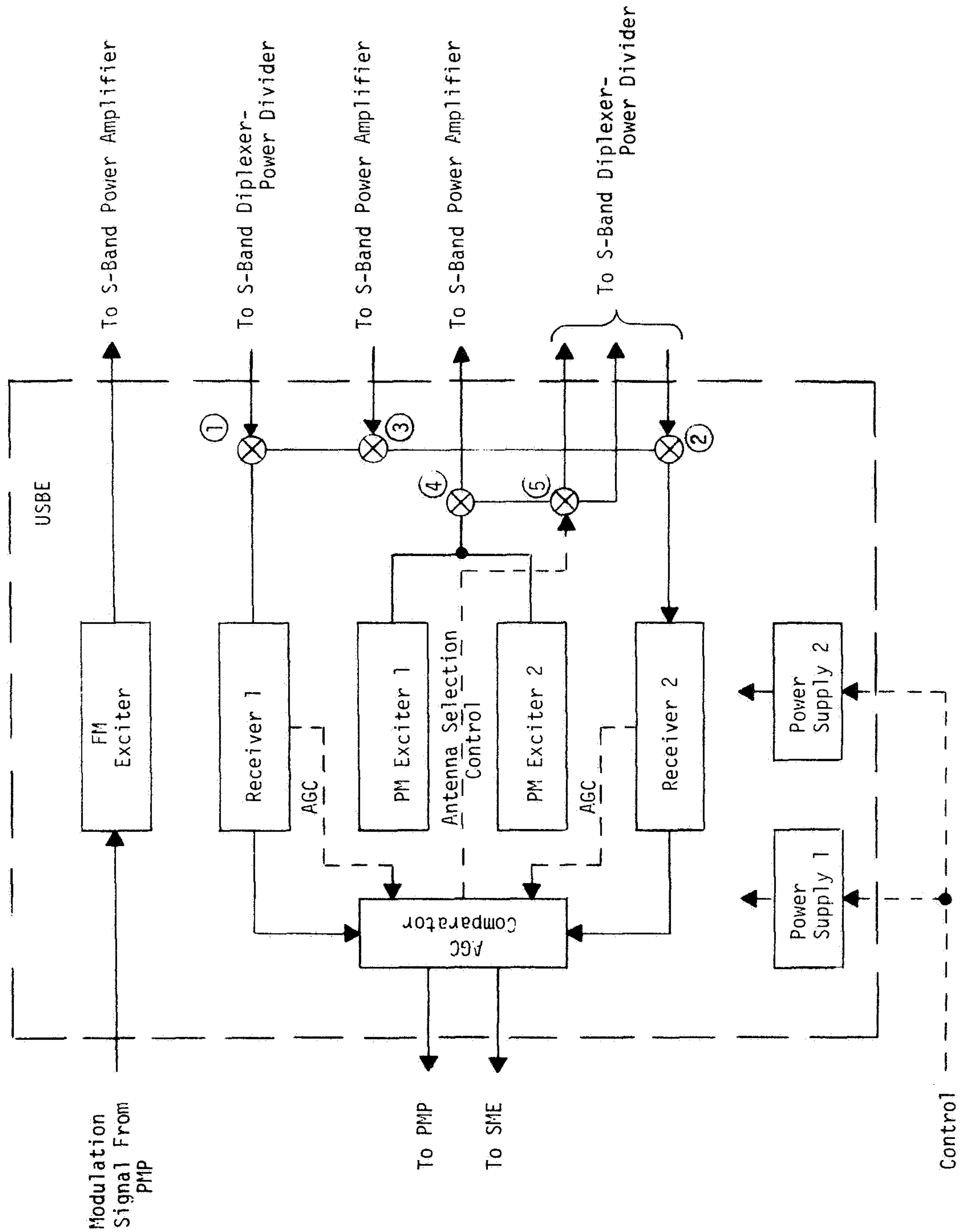


Figure I-1. Modified USBE for ELOR

3. Capability to route each receiver's VCO output to either PM exciter.

4. Capability to sense the AGC of each receiver for selection of the receiver having the best received carrier power level.

5. Capability to select the proper pair of antennas for routing the PM exciter's output based upon the received signal level at the antennas.

6. Independent fail-safe receiver outputs to the PMP and SME.

7. Independent modulation input for SME data subcarrier.

8. Provide all existing modes of operation along with the modifications listed above without significant performance degradation of the unit.

The major items of the modification are the AGC comparator and the RF switches for the receiver inputs and the PM exciter's output. RF switch 3 exists in the present unit configuration and provides the capability to route the received S-Band input signal to receiver 1 or receiver 2. RF switches 1 and 2 are additional requirements which permit the receivers to be connected directly to the S-Band Diplexer-Power Dividers or routed to switch 3 to retain the present capability. RF switch 4 enables the PM exciter's output to be routed to either the S-Band Power Amplifier or to switch 5. RF switch 5 is controlled by the AGC comparator and routes the PM exciter output to the antenna pair receiving the best RF signal.

The AGC comparator employs circuitry to compare the two AGC signals obtained from receiver 1 and receiver 2 for use in selection of receiver output and controlling RF switch 5. The comparator employs a differential amplifier to perform the comparison of the AGC signals. The AGC inputs must be biased to account for the wide RF signal tolerance allowable for a given AGC output from the receiver. The output of the comparator will be a bilevel type signal. When the output changes state, a delay pulse will be generated which will delay receiver output or antenna switching for a short time period. This delay will eliminate the switching of these functions during short term signal drop outs. The analog gates used to route the selected receiver output to the SME will be designed in such a manner that failure of the gate will not affect normal USBE operations.

The modifications will increase the packaged size of the USBE by one inch in depth (new dimensions 9.5 x 6 x 22 inches). The unit weight will increase from 31.6 pounds to approximately 34 pounds. The power required for the USBE for the unattended lunar orbit portion of the mission will be as follows:

Receive Only	10 watts AC, 2 watts DC
Transmit and Receive	26 watts AC, 2.5 watts DC

STATUS MONITORING EQUIPMENT

A block diagram of the status monitoring equipment is presented in Figure 4 of this report. It is to be as follows:

The analog gates, shown in Figure I-2, will sample 40 normalized analog inputs and provide a PAM serial signal to the Analog to Digital (A/D) Converter. In addition to the 40 analog inputs, two reference signals from the SME power supply will be sampled to provide a check on system operation.

The analog gate circuitry will consist of 42 primary analog switches and 11 secondary or sequencer gates. Four primary gates are connected through one secondary gate, preventing the failure of any one gate from affecting more than three other input channels. Drive pulses for the gates will be derived from timing signals from the programmer.

MOSFET switches will be used for all analog switching. These switches were selected because of their enhancement (normally off) mode of operation, which proves a high impedance (open) load on all inputs when power to the gates is off. To minimize the power required, the MOSFET switches will be turned on during only a small portion of the available sample time.

The A/D converter will convert the PAM pulse train from the Analog Gate Circuit to a series of eight bit parallel digital words. The short duration PAM pulses from the Analog Gate Circuit will be stored in a sample and hold circuit for the time required to perform the A/D conversion. A feedback encoder will be used to convert the voltage stored in the sample and hold circuit into an 8 bit parallel digital word. The digital word will then be shifted into the data register for comparison with a preselected upper and lower limit or routed to the output register when transmission of SME data is initiated.

The normal mode of operation for the SME will be as a malfunction detector. In this mode the data register is loaded or preset with the six most significant bits of the data word from the A/D converter. The limit registers are preset with the proper limits, generated from the limit decoding matrix, for the data word being compared. Selection of upper and lower limit reference words for each data input will be by wiring at time of assembly. Changes in reference word after assembly would require rewiring or replacement of the Limit Decoding Matrix. When the three registers are properly loaded, a serial pulse train will be gated into each register. The registers will be counted up from their preset value until each reaches its full state. The register which was preset to the highest count will reach a full state first, the second highest next, and the lowest last.

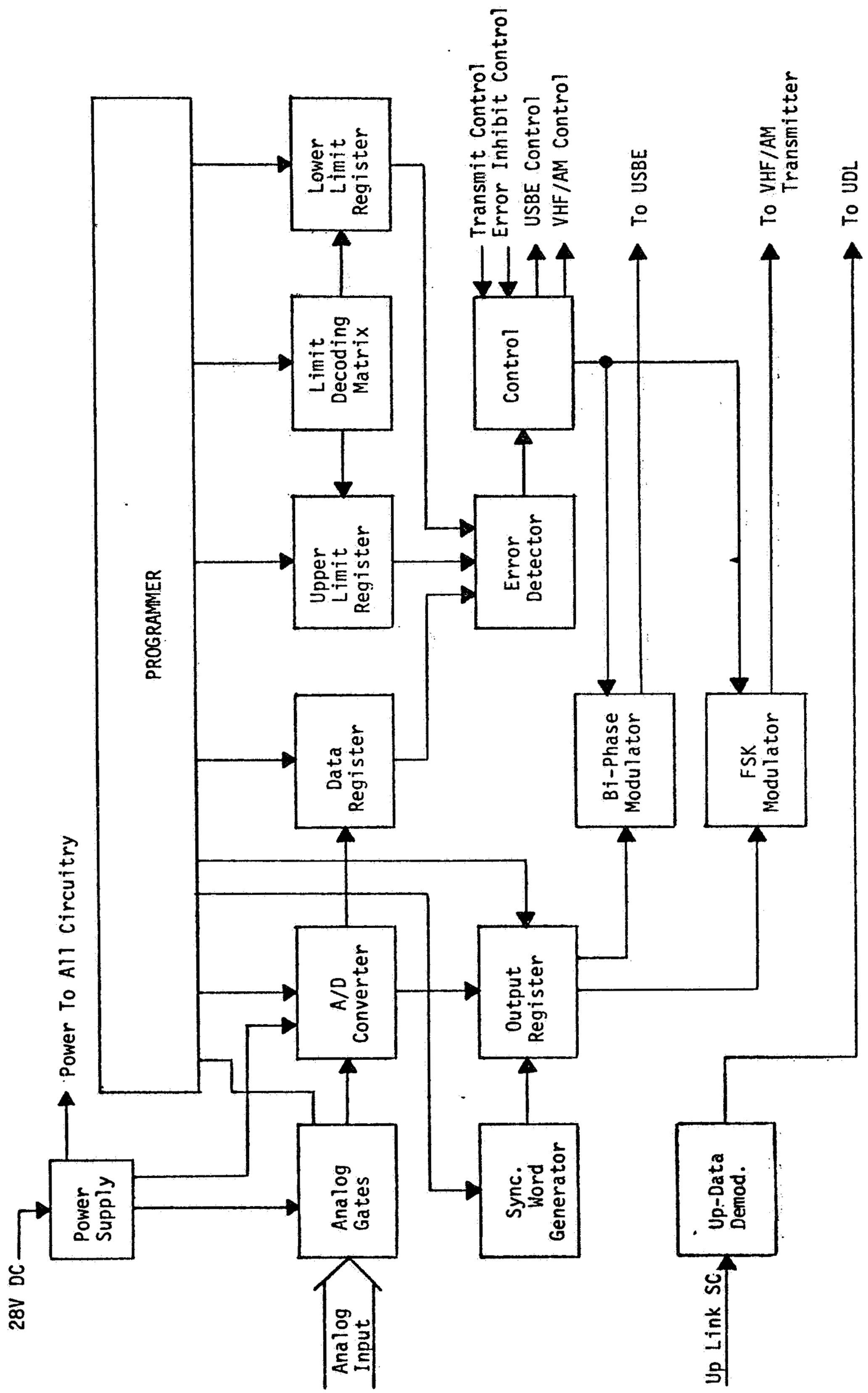


Figure I-2. Block Diagram of Status Monitor

The Error Detector will monitor the outputs of the three registers. The sequence in which the three registers reach their full state will determine if an out-of-tolerance condition exists. An out-of-tolerance condition will result in the generation of an error signal to the control circuit.

The Control circuit will accept the error signal, a Transmit Control or Error Inhibit Control from the Command Demodulator-Decoder or UDL and initiate the proper commands for data transmission. Upon receipt of an error signal, the Bi-Phase Modulator and FSK Generator will be enabled and a contact closure control signal will be transmitted to the USBE and the VHF/AM Transmitter. The transmission of data will continue until the error condition is removed or an Error Inhibit Command is received from the Command Decoder. Data transmission can also be initiated or stopped with a Transmit Command from the Command Decoder. Receipt of the first transmit command will initiate data transmission and receipt of the next command will stop transmission.

During the SME data transmission modes of operation the FSK Modulator receives a serial digital signal while the Bi-Phase Modulator receives a digital signal from the Output Register. The Output Register accepts the 8 bit parallel digital word from the A/D Converter and a 32 bit parallel digital word from the Synchronization Generator to generate its serial output signals.

The Bi-Phase Modulator uses a digital modulation technique of modulo-two addition of the serial digital data with a 4.7 kHz clock. The 4.7 kHz clock is derived from the programmer. After filtering, the output signal from the modulo-two addition circuit provides the 4.7 kHz bi-phase modulated subcarrier used for modulation of the USBE.

The FSK Modulator will generate an 11 kHz tone for one half of each bit period that a "1" is to be transmitted and a 9 kHz tone for one half of each bit period that a "0" is to be transmitted. The transmission of a tone for only one half of each bit period will allow regeneration of a clock signal within the demodulation equipment on the lunar surface/LM with a minimum of circuitry.

The programmer will provide all timing signals required for proper operation of the SME. The programmer will contain a clock oscillator, counter, and decoding circuitry to synchronize the operation of the SME.

A DC to DC converter power supply will provide power to all circuitry within the SME. The power supply will operate from the CSM 28V supply.

In addition to the power required by the SME, three precision voltage outputs will be provided. Two voltages will be provided to the Analog Gate Circuit for SME self test and one precision voltage will be provided to the A/D converter for use in the feedback encoder.

In addition to performing the functions described in section 3.1.1 of this report, the SME houses the Up Data Demodulator. The Up Data Demodulator accepts the output of the USBE's receivers and demodulates the 70 kHz up-data subcarrier. The SME circuitry used to perform this function is similar to the up-data circuitry of the Apollo Block II Premodulation Processor.

The estimated size of the SME is 10" x 8" x 6" (excluding connectors). Estimed SME weight is 18 pounds. These estimates include all reliability redundancy considerations described in section 4 of this report. The estimated power required for the unit is 4 watts from a +28 volt DC source.

COMMAND DEMODULATOR-DECODER

Figure I-3 is a block diagram of the recommended Command Demodulator-Decoder (CDD) required for the proposed lunar surface/LM to CSM remote control link. The demodulator section of the CDD receives a FSK (frequency shift keying) signal from the CSM VHF receiver audio output. The demodulator section detects the FSK signal and provides a digital output consisting of "one" and "zero" bits of the decoded message on separate output lines (RZ format). The demodulator section threshold level is set to provide the digital output when the signal-to-noise ratio at the threshold detectors is high and no output when the signal-to-noise ratio is low. The digital output from the demodulator is free of noise. A very small probability of error in the demodulated output exists, however, as one or more of the demodulator outputs bits could be produced or changed by noise pulses at the threshold detector. The probability of error is made very small by setting a high threshold-to-noise ratio at the threshold detector.

Figure I-4 shows idealized input and output waveforms from the demodulator and the clock signal provided by combining the two demodulator outputs. A "one" bit of the coded message appears at the receiver output as a given frequency tone. A "zero" bit appears as a tone different in frequency from the "one" tone sufficiently separated from the "one" tone to ease the tone filter design. The tones could be the audio range of 300-3000 Hz; but it is preferred to use higher frequencies of 8.6 kHz and 13 kHz, as one of the receiver audio outputs has a specified bandpass of 300-13,000 Hz. The bandpass tone filters of the demodulator separate the "one" and "zero" tones into separate channels. Tone filter bandwidths will be approximately 500 Hz. The narrow bandpass tone filters reject noise except for the noise in the passband and improve the signal-to-noise ratio. The tones are rectified and

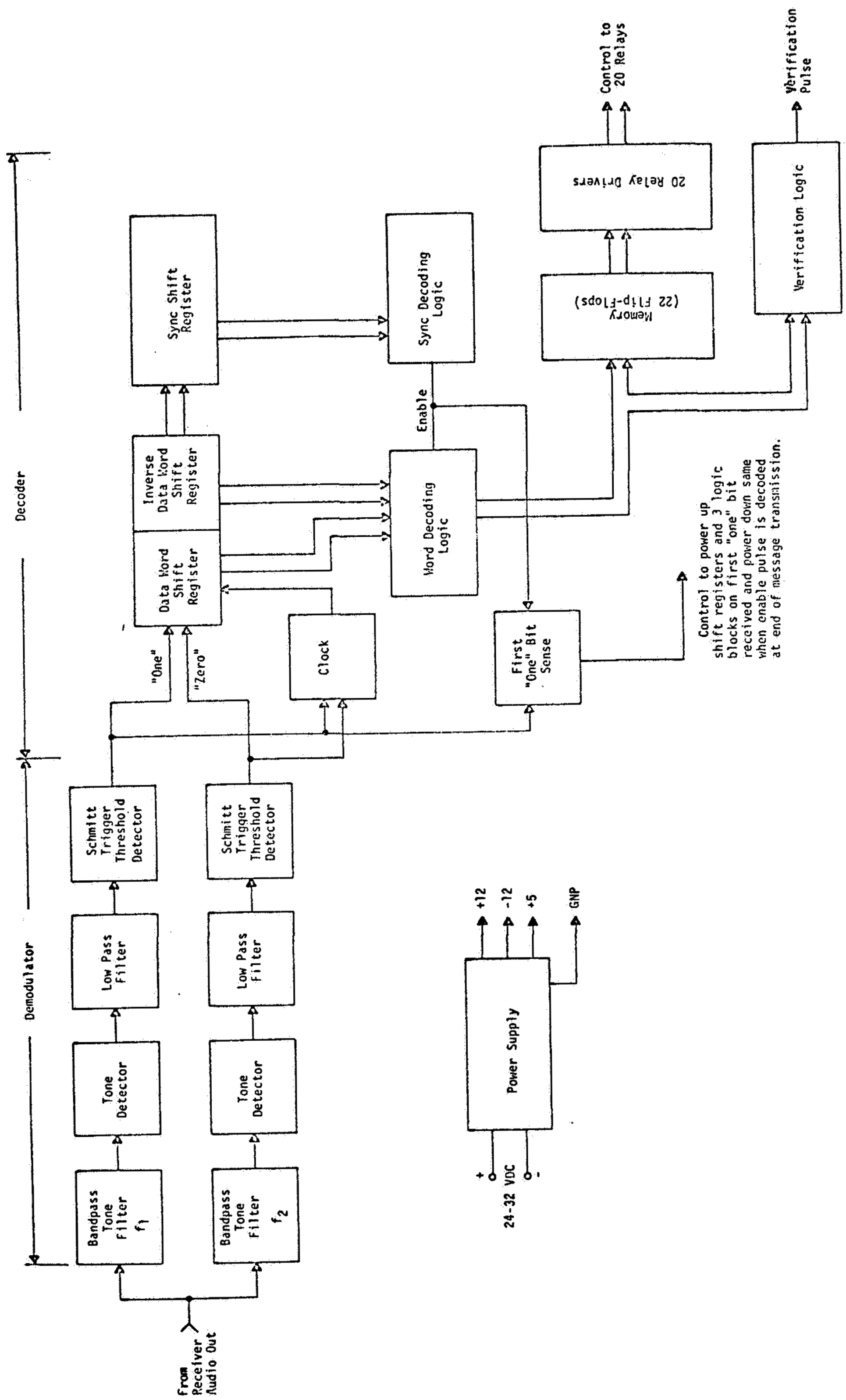
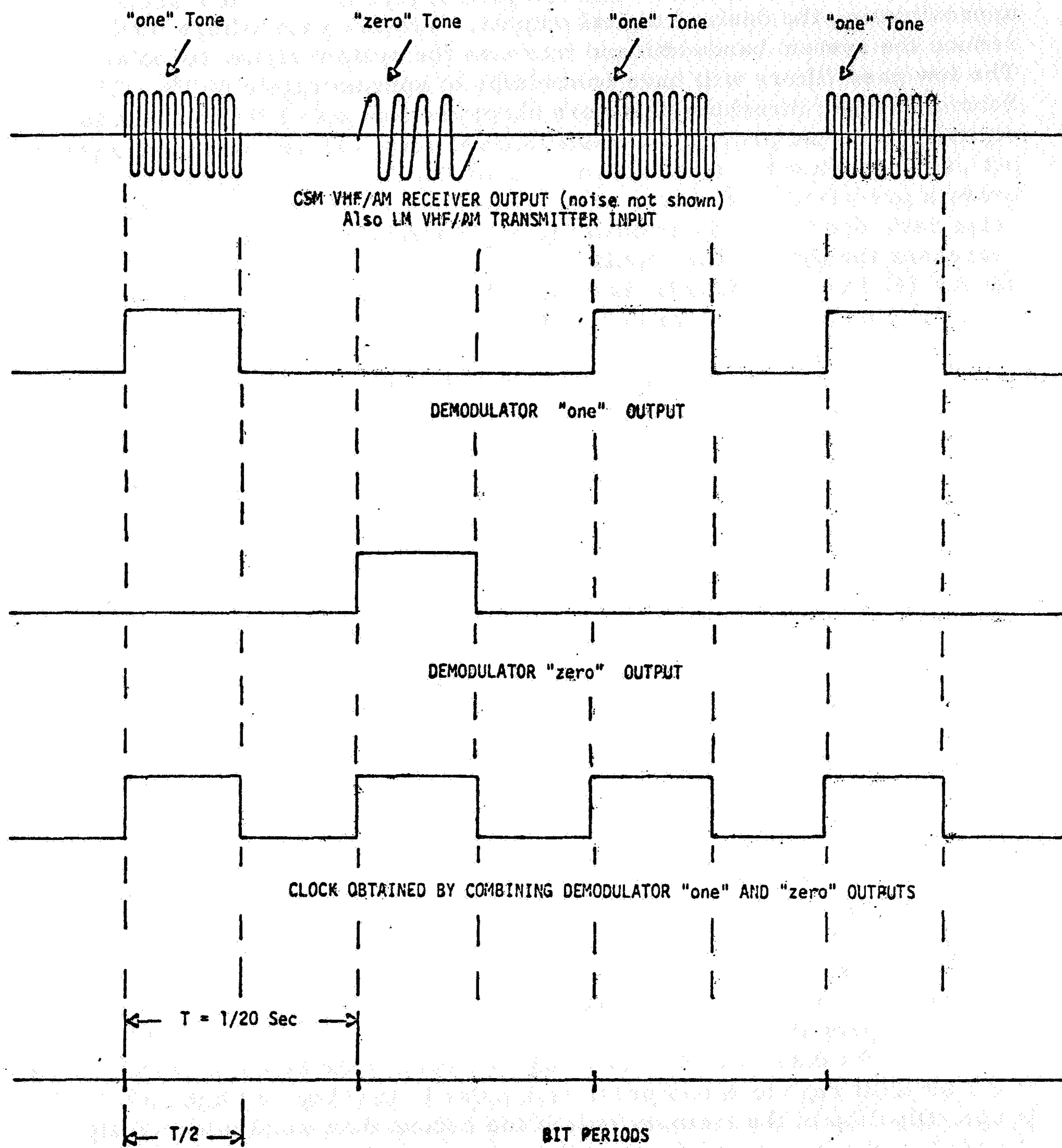


Figure I-3. Block Diagram of Command Demodulator and Decoder



NOTE: Complete message transmission contains 20 bit period. Ten of the bit periods are sync and ten bits for control.

Figure I-4. Idealized Waveform of Small Portion of a Message Transmission filtered by the tone detectors and low pass filters to give wave shapes approximating the desired digital outputs. The low pass filters further reduce the system bandwidth and increase the system signal-to-noise ratio. The low pass filters will have bandwidths of approximately 40 Hz. The Schmitt trigger threshold detectors shape the low pass filter outputs to digital pulses and provide a threshold level such that demodulator output is not provided when the signal-to-noise ratio at the threshold detector falls below a predetermined value. Setting the threshold for a high signal-to-noise ratio decreases the probability of incorrectly receiving a bit but also decreases the system sensitivity. Since the effective CDD bandwidth is narrow (40 Hz), no difficulty is expected in obtaining both the required sensitivity and low error probability.

The modulation approach and demodulation method recommended for the CDD are identical, except for frequencies, to a recent design presently in production at Collins Radio Company. The equipment is the Alert Receiver of the tri-service TACSAT COM (Tactical Satellite Communications) program. The decoder employed in the Alert Receiver is approximately the same as that recommended for the CDD.

A flip-flop at the input to the shift register (not shown on the simplified block diagram) converts the RZ output of the demodulator to NRZ for application to the shift registers. Clock for the shift registers is obtained by an "OR" combination of the "one" and "zero" bits of the RZ outputs from the demodulator. Each high level of the "one" or "zero" demodulator outputs applies a new bit to the shift registers which is clocked into the registers as the input and clock pulses fall to the low level.

The command message length is 20 bits consisting of a 10 bit sync word followed by a 5 bit data word transmitted twice. The 5 bit data word is used to control 20 relay drivers. The data word is transmitted twice, once in normal form and once in inverse form where all "ones" are replaced by "zeros" and all "zeros" are replaced by "ones." To decrease the probability of error, the sync word and both data words must be properly decoded before a command is executed. The sync decoding logic recognizes the sync word pattern and provides an enable pulse whenever the 10 bit sync word is received. The data word decoding logic compares the two data words in the two 5 bit shift registers and provides a pulse to the "toggle" input of the proper flip-flop of the memory only if the second data word is correctly decoded as the inverse of the first data word and the enable pulse is received from the sync decoding logic.

A "toggle" concept is proposed to minimize the number of codes and memory size required to control 20 relays. If separate ON-OFF commands were provided for each relay, the decoding matrix and memory shown in Figure I-4 would be double in complexity, resulting in greater weight as well as a decrease in overall reliability of the CDD. The toggle concept operates in such a fashion that upon receipt of a command the designated relay switches state irrespective of its previous state. If a relay is "on," reception of the relay's corresponding code turns the relay "off"; if the relay is "off," reception of the relay's corresponding code turns the relay "on." A reset code is provided to set all relays, at the same time, to their "off" positions. Thus, the LM encoder knows a reference state for all relays and can keep track of the future states by knowledge of the up-data commands. Multiple commands could be simultaneously executed by use of preset codes. One preset code is provided to switch several relays (as many of the 20 as might be desired) to their "on" positions. With the coding used for the CDD up to ten preset combinations could be provided in addition to the control of the 20 individual relays.

Application of a "toggle" signal to any of the 22 flip-flops in the memory is sensed to provide a verification pulse output. The pulse indicates a valid message has been received and a change of state has been applied through the memory and relay drivers to the proper relay or relays.

Power to the shift registers, word decoding logic, sync decoding logic, and verification logic is applied when the first bit of a message is received and removed after the sync word has been decoded to increase the CDD reliability. In this way seventy-five percent of the decoder is powered down when a message is not being received.

A high efficiency power supply is employed in the Command Demodulator-Decoder to provide +5 VDC for the T154L digital integrated circuits of the decoder and +12 VDC and -12 VDC is provided for the linear integrated circuits of the demodulator section.

Estimated volume of a completed Command Demodulator-Decoder is 140 cubic inches. Estimated weight is 4.3 pounds. Power input is estimated at 5.25 watts during reception of a message and 4.25 watts while monitoring the CSM VHF/AM receiver output. The above estimated powers do not include the power required to operate the 20 relays. It has been assumed that power would be applied to the 20 relay coils from their associated equipments and that the CDD relay drivers would provide grounds for the relay coils to effect "on" conditions and remove the grounds to effect "off" conditions.

S-BAND DIPLEXER-POWER DIVIDER

Two S-Band Diplexer-Power Dividers will be required for the ELOR C&D Subsystem. The diplexer portion of the unit must provide sufficient isolation between the transmit channel at 2287.5 MHz and the receive channel at 2106 MHz. The power divider equally splits the transmit output signal between the pair of S-Band Omni-Directional Antennas.

The unit will be a passive device employing cavity-type filters for the diplexer and a hybrid circuit for the power divider. We have procured several similar type devices for use in our current and past space programs. We estimate that the unit can provide sufficient isolation between the transmit input and receive output within the following insertion loss limits:

Transmit	(Transmit input to diplexer output plus loss in power divider)	1.3 db
Receive	(Single antenna input to output of diplexer)	4.3 db

The estimated size and weight of the unit is 6" x 3" x 1-3/4" and 26 ounces.

APPENDIX J

CSM MEASUREMENT LIST FOR THE APOLLO BLOCK II SPACECRAFT AS APPLICABLE TO ELOR*
(Feb. 1968)

Measurement Description	Data Range		
	Low	High	Units
Temp Side HS Bond Loc 1	-260	+600	°F
Temp Side HS Bond Loc 4	-260	+600	°F
Temp Side HS Bond Loc 7	-260	+600	°F
Temp Side HS Bond Loc 10	-260	+600	°F
Temp Bay 2 Ox Tank Surface	-100	+200	°F
Temp Bay 3 Ox Tank Surface	-100	+200	°F
Temp Bay 5 Fuel Tank Surface	-100	+200	°F
Temp Bay 6 Fuel Tank Surface	-100	+200	°F
Ox Transf Line Entry Sump Tnk	-100	+200	°F
Fuel Transf Line Entry Sump Tnk	-100	+200	°F
Quantity H2 Tank 1	+0	+100	%
Quantity H2 Tank 2	+0	+100	%
Quantity O2 Tank 1	+0	+100	%
Quantity O2 Tank 2	+0	+100	%
Press O2 Tank 1	+50	+1050	psia
Press O2 Tank 2	+50	+1050	psia
Press H2 Tank 1	+0	+350	psia
Press H2 Tank 2	+0	+350	psia
Temp O2 Tank 1	-325	+80	°F
Temp O2 Tank 2	-325	+80	°F
Temp H2 Tank 1	-425	-200	°F
Temp H2 Tank 2	-425	-200	°F
Temp Static Inverter 1	+32	+248	°F
Temp Static Inverter 2	+32	+248	°F
Temp Static Inverter 3	+32	+248	°F
AC Voltage Main Bus 1 Phase A	+0	+150	VAC
AC Voltage Main Bus 2 Phase A	+0	+150	VAC
DC Voltage Main Bus A	+0	+45	VDC
DC Voltage Main Bus B	+0	+45	VDC
DC Voltage Battery Bus A	+0	+45	VDC
DC Voltage Battery Bus A	+0	+45	VDC
DC Voltage Battery Bus B	+0	+45	VDC
DC Voltage Battery Bus B	+0	+45	VDC
DC Current Batt Charger Out	+0	+5	amp

*NR SD data, part of Apollo documentation.

Measurement Description	Data Range		
	Low	High	Units
DC Current Battery A	+0	+100	amp
DC Current Battery A	+0	+100	amp
DC Current Battery A	-3	+100	amp
DC Current Battery B	+0	+100	amp
DC Current Battery B	+0	+100	amp
DC Current Battery B	+0	+100	amp
DC Current Battery C	+0	+100	amp
DC Current Battery C	+0	+100	amp
DC Current Battery C	-3	+100	amp
DC Voltage Battery Relay Bus	+0	+45	VDC
N2 Pressure FC 1 Regulated	+0	+75	psia
N2 Pressure FC 1 Regulated	+0	+75	psia
N2 Pressure FC 2 Regulated	+0	+75	psia
N2 Pressure FC 2 Regulated	+0	+75	psia
N2 Pressure FC 3 Regulated	+0	+75	psia
O2 Pressure FC 1 Regulated	+0	+75	psia
O2 Pressure FC 1 Regulated	+0	+75	psia
O2 Pressure FC 2 Regulated	+0	+75	psia
O2 Pressure FC 2 Regulated	+0	+75	psia
O2 Pressure FC 3 Regulated	+0	+75	psia
O2 Pressure FC 3 Regulated	+0	+75	psia
H2 Pressure FC 1 Regulated	+0	+75	psia
H2 Pressure FC 1 Regulated	+0	+75	psia
H2 Pressure FC 2 Regulated	+0	+75	psia
H2 Pressure FC 2 Regulated	+0	+75	psia
H2 Pressure FC 3 Regulated	+0	+75	psia
H2 Pressure FC 3 Regulated	+0	+75	psia
Temp FC 1 Cond Exhaust	+150	+250	°F
Temp FC 2 Cond Exhaust	+150	+250	°F
Temp EC 3 Cond Exhaust	+150	+250	°F
Temp FC 1 Skin	+80	+550	°F
Temp FC 2 Skin	+80	+550	°F
Temp FC 3 Skin	+80	+550	°F
Temp FC 1 Radiator Outlet	-50	+300	°F
Temp FC 1 Radiator Outlet	-50	+300	°F
Temp FC 2 Radiator Outlet	-50	+300	°F
Temp FC 3 Radiator Outlet	-50	+300	°F
Temp FC 1 Radiator Inlet	-50	+300	°F
Temp FC 2 Radiator Inlet	-50	+300	°F
Temp FC 3 Radiator Inlet	-50	+300	°F
DC Current FC 1 Output	+0	+100	amp
DC Current FC 2 Output	+0	+100	amp

Measurement Description	Data Range		
	Low	High	Units
DC Current FC 3 Output	+0	+100	amp
Flow Rate H2 FC 1	+0	+0.2	lb/hr
Flow Rate H2 FC 2	+0	+0.2	lb/hr
Flow Rate H2 FC 3	+0	+0.2	lb/hr
Flow Rate O2 FC 1	+0	+1.6	lb/hr
Flow Rate O2 FC 2	+0	+1.6	lb/hr
Flow Rate O2 FC 3	+0	+1.6	lb/hr
CSM to LM Current Monitor	+0	+10	amp
DC Voltage Pyro Bus A	+0	+40	VDC
DC Voltage Pyro Bus B	+0	+40	VDC
DC Voltage Logic Bus A	+0	+40	VDC
DC Voltage Logic Bus B	+0	+40	VDC
Pressure Cabin	*0	*17	psia
Temp Cabin	*40	*125	°F
Press O2 Suit to Cabin Diff	*0	*5.2	in H_2O
Press O2 Suit to Cabin Diff	-5	+5	in H_2O
Press CO2 Partial	+0	+30	mm Hg
Press Surge Tank	+50	+1050	psia
Press Surge Tank	+50	+1050	psia
Temp Suit Supply Manif	+20	+95	°F
Quantity Waste Water Tank	+0	+100	%
Quan Potable H20 Tank	+0	+100	%
Press Suit Demand Reg Sense	+0	+17	psia
Press Suit Compressor Diff	+0	+1	psid
Press Suit Compressor Diff	+0	+1	psid
Press Glycol Pump Outlet	+0	+60	psig
Temp Glycol Evap Outlet Steam	+20	+95	°F
Temp Glycol Evap Outlet Liquid	+25	+75	°F
Quantity Glycol Accum	+0	+100	%
Temp Space Radiator Outlet	-50	+100	°F
Back Press Glycol Evaporator	+0.05	+0.25	psia
Back Press Glycol Evaporator	+0.05	+0.25	psia
Flowrate ECS O2	+0.2	+1.0	lb/hr
Flowrate ECS O2	+0.2	+1.0	lb/hr
Press Outlet O2 Reg Supply	+0	+150	psig
Press Outlet O2 Reg Supply	+0	+150	psig
Press Secondary Glycol Pump	+0	+60	psig
Press Secondary Glycol Pump	+0	+60	psig
Temp Secondary Evap Out Liquid	+25	+75	°F
Temp Secondary Evap Out Liquid	+25	+75	°F
Quantity Secondary Glycol Accum	+0	+100	%
Quantity Secondary Glycol Accum	+0	+100	%

Measurement Description	Data Range		
	Low	High	Units
Press Secondary Evap Out Steam	+0.05	+0.25	psia
Press Secondary Evap Out Steam	+0.05	+0.25	psia
Pressure Water and Glycol Tanks	0	+50	psia
Rate Glycol PRI Coldplate Outlet	+150	+300	lb/hr
Temp Glycol Evaporator Inlet	+35	+100	°F
Temp Primary Radiator Inlet	+55	+120	°F
Temp Secondary Radiator Inlet	+55	+120	°F
Temp Secondary Radiator Outlet	+30	+70	°F
Temp Waste Water Dump Nozzle	+0	+100	°F
Temp Urine Dump Nozzle	0	+100	°F
+120 VDC PIPA Supply DC Level	+85	+135	VDC
2.5 VDC TM Bias	0	+5	VDC
IMU 28V .8KC 1 PCT 0 Deg RMS	0	31.1	VRM
3.2KC 28V Supply	0	31.1	VRM
IG 1X Resolver Output Sin	0	360	°F
IG 1X Resolver Output Cos	0	360	°F
IGA Servo Error in Phase	-3.0	+3.0	VRMS
MG 1X Resolver Output Sin	0	360	°F
MG 1X Resolver Output Cos	0	360	°F
MGA Servo Error in Phase	-3.0	+3.0	VRMS
OG 1X Resolver Output Sin	0	360	°F
OG 1X Resolver Output Cos	0	360	°F
OGA Servo Error in Phase	-3.0	+3.0	VRMS
PIPA Temperature	+120	+140	°F
Shaft CDU DAC Output	-10	+10	VRMS
Trunnion CDU DAC Output	-10	+10	VRMS
FDAI CMC/SCS Att Error Pitch	OR	+5/5/15	°F
FDAI CMC/SCS Att Error Yaw	OR	+5/5/15	°F
FDAI CMC/SCS Att Error Roll	OR	+5/5/50	°F
FDAI SCS Body Rate Pitch	OR	+1/5/10	deg/sec
FDAI SCS Body Rate Yaw	OR	+1/5/10	deg/sec
FDAI SCS Body Rate Roll	OR	+1/5/50	deg/sec
Gimbal Position Pitch 1 or 2	-5.0	+5.0	°F
Gimbal Position Yaw 1 or 2	-5.0	+5.0	°F
SCS TVC Auto Command Pitch	-10	+10	VDC
SCS TVC Auto Command Yaw	-10	+10	VDC
Rotational Control/MTVC Pitch Cmd	-11.5	+11.5	°F
Rotational Control/MTVC Yaw Cmd	-11.5	+11.5	°F
Rotational Controller Roll Cmd	-11.5	+11.5	°F
TVC Pitch Diff Current	0.835	+0.835	amp
TVC Yaw Diff Current	0.835	+0.835	amp
EKG Commander LH Couch	+0.1	+5	mv

Measurement Description	Data Range		
	Low	High	Units
EKG Cmd Module Pilot CTR Couch	+0.1	+5	mv
EKG Lunar Module Pilot RH Couch	+0.1	+5	mv
Resp Rate Commander LH Couch	-5	+5	ohm
Resp Rate CM Pilot CTR Couch	-5	+5	ohm
Resp Rate LM Pilot RH Couch	-5	+5	ohm
CM Accelerometer X-Axis	-2	+10	G
CM Accelerometer X-Axis	-2	+2	G
CM Accelerometer	-2	+2	G
Radiation Dosimeter 1	+0	+1K	rad/hr
Radiation Dosimeter 2	+0	+1K	rad/hr
Dosimeter Rate Change	+0	+5	VDC
Scientific Experiment No. 1			
Scientific Experiment No. 2			
Scientific Experiment No. 3			
Scientific Experiment No. 4			
Scientific Experiment No. 5			
Scientific Experiment No. 6			
Scientific Experiment No. 7			
Scientific Experiment No. 8			
Scientific Experiment No. 9			
Scientific Experiment No. 10			
Scientific Experiment No. 11			
Scientific Experiment No. 12			
Scientific Experiment No. 13			
Scientific Experiment No. 14			
Scientific Experiment No. 15			
He Press Tank	+0	+5K	psia
He Temp Tank	-100	+200	°F
Press Oxidizer Tanks	+0	+250	psia
Press Oxidizer Tanks	+0	+250	psia
Press Fuel Tanks	+0	+250	psia
Press Fuel Tanks	+0	+250	psia
Position Fuel/Ox VLV 1 Pot B	+0	+90	°F
Position Fuel/Ox VLV 2 Pot B	+0	+90	°F
Position Fuel/Ox VLV 3 Pot B	+0	+90	°F
Position Fuel/Ox VLV 4 Pot B	+0	+90	°F
Temp Engine Valve Body	+0	+200	°F
Temp Engine Fuel Feed Line	+0	+200	°F
Temp Engine Oxidizer Feed Line	+0	+200	°F
Temp Engine Oxidizer Feed Line	+0	+200	°F
Temp 1 Ox Distribution Line	+0	+200	°F
Temp 1 Fuel Distribution Line	+0	+200	°F

Measurement Description	Data Range		
	Low	High	Units
Eng Injector Flange Temp No. 1	+0	+500	°F
Eng Injector Flange Temp No. 2	+0	+500	°F
Eng VLV Act Sys Tank Press PRI	+0	+5K	psia
Eng VLV Act Sys Tank Press SEC	+0	+5K	psia
Quan Ox Tank 1 PRI - Total Aux	+0	+50	%
Quan Ox Tank 2	+0	+60	%
Quan Fuel Tank 1 PRI - Total Aux	+0	+50	%
Quan Fuel Tank 2	+0	+60	%
Press Engine Chamber	+0	+150	psia
Press Fuel Sm/Eng Interface	+0	+300	psia
Press Ox Sm/Eng Interface	+0	+300	psia
A He Manifold Press	+0	+400	psia
A He Manifold Press	+0	+400	psia
A He Manifold Press	+0	+400	psia
Oxidizer Manifold Press Sys A	+0	+300	psia
Oxidizer Manifold Press Sys A	+0	+300	psia
Fuel Manifold Press Sys A	+0	+300	psia
Fuel Manifold Press Sys A	+0	+400	psia
B He Manifold Press	+0	+400	psia
B He Manifold Press	+0	+400	psia
B He Manifold Press	+0	+400	psia
Oxidizer Manifold Press Sys B	+0	+300	psia
Oxidizer Manifold Press Sys B	+0	+300	psia
Fuel Manifold Press Sys B	+0	+300	psia
Fuel Manifold Press Sys B	+0	+400	psia
C He Manifold Press	+0	+400	psia
C He Manifold Press	+0	+400	psia
C He Manifold Press	+0	+400	psia
Oxidizer Manifold Press Sys C	+0	+300	psia
Oxidizer Manifold Press Sys C	+0	+300	psia
Oxidizer Manifold Press Sys D	+0	+300	psia
Oxidizer Manifold Press Sys D	+0	+300	psia
Fuel Manifold Press Sys C	+0	+300	psia
Fuel Manifold Press Sys C	+0	+400	psia
Fuel Manifold Press Sys D	+0	+300	psia
Fuel Manifold Press Sys D	+0	+400	psia
D He Manifold Press	+0	+400	psia
D He Manifold Press	+0	+400	psia
D He Manifold Press	+0	+400	psia
Angle of Attack	0	+5	VDC
Temp Docking Probe Cylinder	-100	+250	°F
Sig Cond Pos Supply Volts	+0	+30	VDC

Measurement Description	Data Range		
	Low	High	Units
Sig Cond Neg Supply Volts	+0	-30	VDC
Sensor Excitation 5 Volts	+0	+9	VDC
Sensor Excitation 10 Volts	+0	+15	VDC
PCM Hi Level 85 Percent Ref	+0	+5.0	VDC
PCM Hi Level 15 Percent Ref	+0	+1.0	VDC
S-Band Rec 1-2 AGC Voltage	-130	-50	dbm
S-Band RCVR 1-2 Static PH Error	-90	+90	kHz
Proton Count Rate Channel 1	+1	+100K	C/S
Proton Count Rate Channel 2	+0.1	+10K	C/S
Proton Count Rate Channel 3	+0.1	+10K	C/S
Proton Count Rate Channel 4	+0.1	+10K	C/S
Alpha Count Rate Channel 1	+0.1	+10K	C/S
Alpha Count Rate Channel 2	+0.1	+10K	C/S
Alpha Count Rate Channel 3	+0.1	+10K	C/S
Proton Inter Count Rate	+1	+100K	C/S
Temp Nuclear Particle Detector	-65	+140	°F
Temp Nuclear Particle Analyzer	-65	+140	°F
P He Press Tank A	+0	+5K	psia
P He Press Tank 1	+0	+5K	psia
P He Press Tank B	+0	+5K	psia
P He Press Tank 2	+0	+5K	psia
T He Temp Tank A	+0	+300	°F
T He Temp Tank 1	+0	+300	°F
T He Temp Tank 1	+0	+300	°F
T He Temp Tank B	+0	+300	°F
T He Temp Tank 2	+0	+300	°F
T He Temp Tank 2	+0	+300	°F
P Press Fuel Tank A	+0	+400	psia
P Press Fuel Tank	+0	+400	psia
P Press Fuel Tank A	+0	+400	psia
P Press Fuel Tank B	+0	+400	psia
P Press Fuel Tank 2	+0	+400	psia
P Press Fuel Tank B	+0	+400	psia
P Press Oxidizer Tank A	+0	+400	psia
P Press Oxidizer Tank 1	+0	+400	psia
P Press Oxidizer Tank A	+0	+400	psia
P Press Oxidizer Tank B	+0	+400	psia
P Press Oxidizer Tank 2	+0	+400	psia
P Press Oxidizer Tank B	+0	+400	psia
P Press CM-RCS Helium Manifold 1	0	+400	psia
P Press CM-RCS Helium Manifold 2	0	+400	psia
P He Press Tank A	+0	+5K	psia

Measurement Description	Data Range		
	Low	High	Units
P He Press Tank B	+0	+5K	psia
P He Press Tank C	+0	+5K	psia
P He Press Tank D	+0	+5K	psia
T He Temp Tank A	-100	+200	°F
T He Temp Tank B	-100	+200	°F
T He Temp Tank C	-100	+200	°F
T He Temp Tank D	-100	+200	°F
T He Temp Tank A	+0	+100	°F
T He Temp Tank A	+0	+100	°F
T He Temp Tank B	+0	+100	°F
T He Temp Tank B	+0	+100	°F
T He Temp Tank C	+0	+100	°F
T He Temp Tank C	+0	+100	°F
T He Temp Tank C	+0	+100	°F
T He Temp Tank D	+0	+100	°F
T He Temp Tank D	+0	+100	°F
Q Quantity SM RCS Propellant Sys A	+0	+100	%
Q Quantity SM RCS Propellant Sys B	+0	+100	%
Q Quantity SM RCS Propellant Sys C	+0	+100	%
Q Quantity SM RCS Propellant Sys D	+0	+100	%
T Temp Engine Package A	+0	+300	°F
T Temp Engine Package B	+0	+300	°F
T Temp Engine Package C	+0	+300	°F
T Temp Engine Package D	+0	+300	°F
T Temp Engine Package D	+0	+300	°F

APPENDIX K

PROPOSED ALLIS CHALMERS CSM FUEL CELL POWER PLANT FOR EXTENDED LUNAR ORBITAL RENDEZVOUS MISSIONS*

CSM FUEL CELL DESCRIPTION

The ELOR power plant subsystem design considered for the orbiting service module utilizes the capillary matrix 2 kw fuel cell module which has been chosen for the Apollo Applications Program and which is now undergoing 1500 hour qualification. This module design has demonstrated a life capability of 3000 hours which is greater than the ELOR requirements.

The module has a gas cooled (He) fuel cell stack consisting of 31 sections which operates at 190°F. Vehicle supplied coolant circulates through tubes wrapped around the module canister removing the heat from the helium gas with the heat being dissipated in the vehicle radiator. Water removal is accomplished by means of the static moisture removal concept. The moisture is vented directly to space during the unmanned portion of the lunar orbit.

Module operation illustrating the design for AAP is shown by the schematic (Figure K-1). Changes to this design for ELOR applications will be discussed in following paragraphs. This configuration is also applicable to AAP qualification program, except that the fan and coolant pump inverter package, together with its cold plate, will be eliminated. Power for the fans and coolant pump is provided by the vehicle inverter. The graph in Figure K-2 shows the output characteristics of one module. The lower curve represents module characteristics at the end of 2160 hours projected from the 1500 hour curve using the performance requirements for the 1500 hour qualification program. Module heat transferred to the vehicle coolant is accomplished through a dual loop process where helium is the transfer medium to the vehicle system coolant loop. Since potable water is not required during nearly the entire duration of the lunar orbit, the lower open-loop curves are applicable. Oxygen consumption for a single module throughout the mission duration is nearly a linear function of gross power as is the hydrogen consumption.

The following outlines the module design by discussing the major module assemblies together with the changes necessary to make it applicable to the ELOR requirements. Each assembly with its elements, together with the required design changes, is summarized in Table K-1.

*Provided by the Allis Chalmers Div. of G.M. through Reference 4.17 of Vol. I.

Table K-1. DVT Fuel Cell System Elements Revised per ELOR Requirements

Item	Remarks
Reactant Conditioning Assembly	
Inlet Solenoid Valves, H_2, O_2.	Latching (Vehicle Equipment)
Preheaters, H_2, O_2	
Dual Pressure Regulator	
Over-Pressure Switches, H_2, O_2	
Pressure Transducers, H_2, O_2	
Fluid Interface Connectors	No Change
Electrical Interface Connectors	
Tubing	
Wiring	
Electrical Component Connectors	
Purge Assembly	
Purge Valves, H_2, O_2	
Outlet Restrictors, H_2, O_2	
Fluid Interface Connectors	
Electrical Interface Connectors	No Change
Tubing	
Wiring	
Electrical Component Connectors	
Temperature Conditioning Assembly	
Cannister Fill Valve, He	Add He make-up tank and controls
Pressure Transducer, He	
Thermistor, Heat and Reactant Control	
Heater (2), 800 W	Add low wattage standby heaters
Thermostat (2), Heater Temp. Limiting	
Over-Temp. Switch	
Fans (2), He	Add switching control
Temperature Sensors, Platinum	
Ducting	

Table K-1. DVT Fuel Cell System Elements
Revised per ELOR Requirements (Cont)

Item	Remarks
Temperature Conditioning Assembly (Cont)	
Fluid Interface Connectors	
Electrical Interface Connectors	
Tubing	
Wiring	
Electrical Component Connectors	
Liquid Coolant Assembly	
Filter	
Relief Valve	
Pump	Second parallel pump added and switching controls
Cold Plate, Inverter	Eliminate
Heat Exchanger, Condenser	By-passed during Lunar Orbit
Valve, Relief	
Valve, Solenoid	
Heat Exchanger, Canister	
Accumulator	
Fluid Interface Connectors	
Electrical Interface Connectors	
Tubing	
Wiring	
Electrical Component Connectors	
Moisture Removal Assembly	
Pressure Transducer, Water Vapor, Readout	
Pressure Transducer, Water Vapor, Control	
Temp. Sensor, Stack, Control	
Valves, Moisture Removal (2)	No Change
Electrical Interface Connectors	
Tubing	
Wiring	
Electrical Component Connectors	

Table K-1. DVT Fuel Cell System Elements
Revised per ELOR Requirements (Cont)

Item	Remarks
Water Recovery Assembly	
Valve, Condenser By-Pass	In by-pass mode during Lunar Orbit
Pressure Switch, Condenser	Not Operating
Condenser	Not Used
Deionizer	Not Used
Check Valve, Pump Inlet	Not Operating
Pressure Regulator, O_2 and Stop Valve	Closed
Relief Valve, O_2	Not Needed No Design Change
Valve, Pump Control	Not Used
Vacuum Regulator, O_2 Vent	Not Used
Vacuum Regulator, Water Vent	Not Used
Fluid Interface Connectors	
Tubing	
Wiring	
Electrical Component Connectors	
EMCA	Add remote startup switching control - Change packaging design
Bus, 28 VDC, EMCA Power, and Steering Diodes, Suppression CKT	
Bus, 28 VDC, Valve/Relay Power and Steering Diodes	
Terminal Board	
Bus, Ground	
Heater, and Reactant Controller	Used during startup only
Voltage Regulator	
Temp. Comparator, Heater Control	
Temp. Comparator, Reactant Valve	
Reactant Valve Control	

Table K-1. DVT Fuel Cell System Elements Revised per ELOR Requirements (Cont)

Item	Remarks
EMCA (Cont)	
Purge Control	
Voltage Regulator	
Ampere Hour Counter	
Purge Valve Control	
Coolant Control	
Voltage Regulator	
Temp. Comparator	Modify for use with latching valve
Coolant Valve Control	
Cavity Pressure Control	
Voltage Regulator	
Comparator, Press., Temp.	No circuit design change
Moisture Removal Valve Cont.	
Water Recovery Control	Not used during Lunar Orbit
Voltage Regulator	
By-pass Valve Control, Condense	
By-pass Valve Control, Vent	
Pump Valve Control	
Wiring	
Electrical Component Connectors	
Inverter, Fan (2)	Eliminated - Add switching circuit to turn on fans
Voltage Regulator	
Square Wave Generator	
Shaper	
Transformer	
Inverter, Coolant Pump	Eliminated - Add switching circuit to turn on pump
Voltage Regulator	
Square Wave Generator	
Shaper	
Transformer	

Table K-1. DVT Fuel Cell System Elements
Revised per ELOR Requirements (Cont)

Item	Remarks
Stack Assembly	
Stack, 31 Sections	
Current Shunt	
Canister End Plate	
Canister	
Canister Dome	
Frame	Change to Reflect Assembly Rearrangements
Insulation	Change in Shape
Fluid Connectors	
Electrical Interface Connectors	
Tubing	
Wiring	
Electrical Connectors	

REACTANT CONDITIONING ASSEMBLY

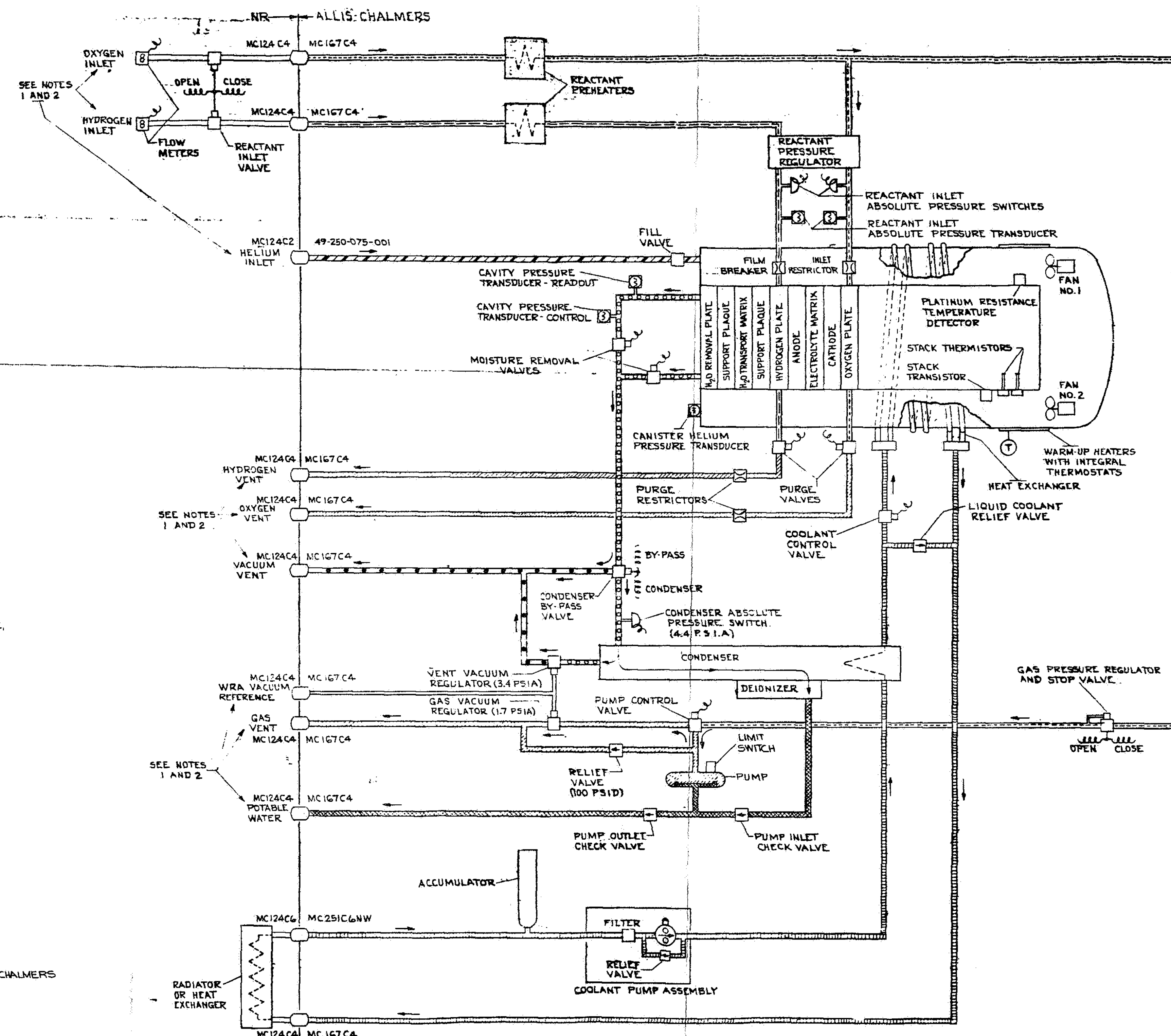
Reactants are supplied to each module through vehicle provided latching valves. Reactant pressure is reduced and balanced by the dual pressure regulator on the fuel cell module. An overpressure condition initiates closing of the reactant valves by means of pressure switches. Pressure transducers provide monitoring of the reactant pressures. No changes in design are anticipated for ELOR requirements. However, maintaining a module in the standby condition for ninety days requires that the reactant valves be closed to eliminate possible degradation of the cells.

PURGE ASSEMBLY

Reactant impurities are removed from the stack manifold and vented to space by means of a solenoid valve in each of the reactant vent lines. The valves are opened when a preset quantity of charge (ampere-hours) has been delivered by the module. Line restrictors control reactant flow during a purge. No design changes are anticipated for ELOR.

TEMPERATURE CONDITIONING ASSEMBLY

The stack coolant, helium, transfers heat to and from the cell fins and the canister wall by means of a fan. The canister wall is heated during startup by two 800 watt heaters powered by ground support equipment. During module operation, the wall is cooled by vehicle coolant. During



NOTES

1. FOR REQUIREMENTS OF FLOW, PURITY PRESSURE, TEMPERATURE, ETC. REFER TO QUAL. SYSTEM PERFORMANCE SPECIFICATION

LEGEND

DESIGNATES AN INTERFACE - NR/ALLIS-CHALMERS
HYDROGEN
OXYGEN
HELIUM
COOLANT (GLYCOL/WATER
POTABLE WATER
HYDROGEN VENT
OXYGEN VENT
STEAM
STEAM VENT

Figure K-1. Fuel Cell Power Plant Schematic

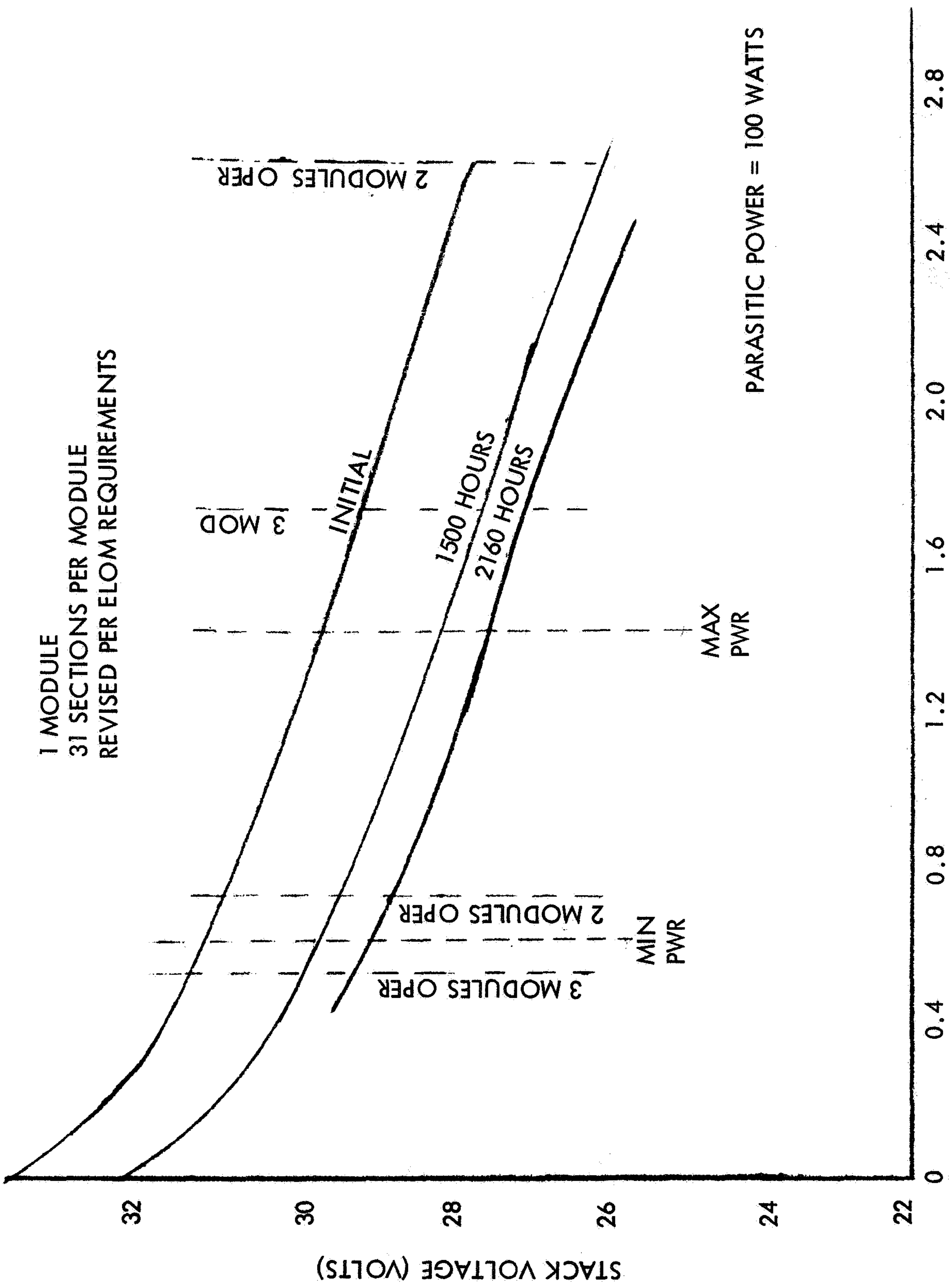


Figure K-2. Voltage Versus Gross Power

ground startup heater power is controlled by the ground support equipment in response to a module signal developed by the thermistor imbedded in the stack. Heater surface temperature limiting is provided by a thermostat imbed in each heater. Platinum resistance temperature sensors are used for stack temperature monitoring. A pressure transducer provides helium pressure information for monitoring. The initial helium charge is introduced into the canister before vehicle installation by means of a manually operated valve. The design changes anticipated for ELOR are the following:

1. Addition of a helium make-up tank for the entire power plant together with pressure controls for each module to provide added helium to replace that which may be lost through diffusion or leakage during the 90 day-plus mission.

2. An additional heater is required for the hot standby module. This allows immediate use of the standby module if the operating module fails. The "bootstrap" startup procedure will be employed should it become necessary to immediately start the cold standby module.

LIQUID COOLANT ASSEMBLY

Vehicle coolant is circulated through the canister coils, the heat exchanger and the cold plate by means of the module coolant pump. The coolant control solenoid valve (normally closed) closes when the stack temperature, sensed by a thermistor, drops to a predetermined level. The coolant by-passes the canister heat exchanger through the relief valve, maintaining the stack at its design temperature. The accumulator compensates for the volume changes in the coolant due to temperature changes. The following modifications to this assembly are anticipated for ELOR:

1. A second coolant pump is added in parallel with the first two reasons; first, to increase coolant flow to the canister heat exchanger so that a higher coolant inlet temperature than in the earlier designs can be employed. Second, the added pump is required only during high loading of the module and would not be energized during the lunar orbit, thereby increasing the reliability of the system.

2. Since AC power is provided by vehicle equipment, the inverter and its cold plate is not required.

3. During the lunar orbit, since no potable water is required, the product water by-passes the condenser and no heat is contributed to the coolant by the condenser.

4. To keep the cold standby modules from freezing, the coolant pump must be operated, and radiator loop must be thermally (not liquid) interconnected. The pump power (approx. 30 watts per pump) can (should) form a part of the vehicle total load being used to keep the vehicle radiator from freezing.

5. The coolant by-pass solenoid valve is to be changed to a latching valve. This provides two advantages: first, during the standby mode, the valve will remain open without drawing parasitic power. Second, the valve will have a lower failure rate because of the reduced duty cycle.

MOISTURE REMOVAL ASSEMBLY

The flow of water vapor from the stack is controlled by regulating the water cavity pressure by means of the two moisture removal solenoid valves. The water vapor pressure transducer and stack temperature sensor outputs are combined in the EMCA water cavity controller so that the valve operation produces a constant electrolyte concentration in the stack. The moisture is then either sent to the condenser or vented to space. Monitoring of the water vapor pressure is accomplished by the use of a second pressure transducer. No changes in design or operation are necessary for ELOR.

WATER RECOVERY ASSEMBLY

The condenser by-pass valve sends the stack water vapor to the condenser or to space depending on the position of the condenser pressure switch. When a decrease in water usage causes the condenser vent to increase in pressure, the pressure switch causes the by-pass valve to vent water vapor to space. Condensed water flows through a deionizer to assure an acceptable pH and is sent to the diaphragm pump. When the diaphragm is filled, a limit switch is activated causing the pump control valve to latch for a short time in its pumping position. This position allows oxygen to enter the cavity on the gas side of the diaphragm forcing the water into the potable water storage assembly. The valve then returns to its original position and the pumping cycle is repeated. No design changes are anticipated for ELOR. However, during lunar orbit this assembly will not be operating since potable water is not required. The condenser by-pass valve will be manually switched to the vent position and vents the water to space.

ELECTRICAL MONITORING AND CONTROL ASSEMBLY (EMCA)

The EMCA is an electronic assembly containing the following controls:

Ground Startup - When the fuel cell module is provided a start signal, and the fuel cell stack is below operating temperature, as sensed by the stack thermistor, a signal is provided to the vehicle ground support interface calling for power to the startup heaters. When the stack reaches near-operating temperature, this signal is turned off, a "ready" signal is provided to the instrument panel and a command is sent to the vehicle interface to allow automatic reactant valve opening. No design changes of this control are anticipated for the ELOR mission. This control is required to operate only during module startup and can be physically separated from the rest of the EMCA and placed with other controls whose usage rate is low since they will probably not require spares.

In-Space Hot Standby - Standby heaters and associated controls must be added to maintain the module near operating temperature such that full power will be available if the operating module fails.

In-Space Cold Standby - No changes to the EMCA are required. If a module must be started while in this mode, bootstrap startup techniques will be employed to bring itself up to operating temperature and this requires no additional equipment.

PURGE CONTROL

The purge control receives a current signal from a shunt in the module assembly and integrates it with respect to time. When a preset ampere-hour quantity is reached, the purge valves are opened for a preset time period. No design change is anticipated for this control for ELOR application. This control, however, since it has a high usage rate and greater complexity, may require a spare and thus should be considered for a separate assembly. This would require repackaging of the EMCA into at least two assemblies which could be replaceable under EVA conditions.

COOLANT CONTROL

This circuit receives temperature information from the stack thermistor and energizes (opens) the coolant control valve when the temperature rises to a preset value. Changes in this circuit for ELOR include modifications to be compatible with a solenoid latching valve. This control is continuously active for an operating module and could be repackaged similar to the purge control.

CAVITY PRESSURE CONTROL

This circuit regulates the vapor pressure in the water cavity by activating the moisture removal valves in response to the pressure transducer and stack transistor temperature signals. In addition, a signal is provided to the instrument panel that a valve command is being generated. No circuit design changes are necessary for ELOR. A packaging change similar to that of the purge control is considered applicable.

INVERTER

In the module designed for DVT, separate inverters supply three-phase, 400 Hz power for the two fans and coolant pump. In ELOR, as in AAP, AC power will be provided by the vehicle. It is necessary only to provide switching circuits to control startup and shutdown of these motors.

STACK ASSEMBLY

The stack assembly consists of a 31 section fuel cell stack housed in the canister assembly. The canister assembly is mounted on a frame and covered with an insulation blanket. Changes reflecting the ELOR requirements are modifications of the frame to allow access to the replaceable items, and quick-adaptation for ISM hardware, and changes to the insulation blanket.

APPENDIX L

FUEL CELL SYSTEM FAILURE RATE ESTIMATE*

Table L-1. Fuel Cell System Failure Rate Estimate
(Based on Allis Chalmers System)

Part or Component Description	Quantity	λ	Total Quantity x λ
FC Stack			
Fuel Cells	31	.010	
Seals, Stack	248	.014	
End Plate	10	.014	
Canister	4	.014	
			3.978
RCC (Reactant Conditioning)			
Dual Pressure Reg.	1	2.000	
Purge Valve	2	1.000	
Restrictors	2	.330	
Pressure Switch	2	.500	
Transducer	2	2.500	
Fittings		1.200	
			5.030
TCC (Temperature Conditioning)			
Fans	2	30.000	
Coolant Valve	1	1.000	
By-Pass Valve	1	1.000	
Heater	1	.025	
Pump	1	5.000	
Accumulator	1	.350	
Heater	1	.700	
Thermostat-II	1	.300	
Thermostat Stack	1	.300	
Temp Detector	1	.300	
Transducer	1	2.500	
Fittings		1.000	
			4.375

*Provided by Allis Chalmers Div. of GM through Reference 4.17 of Vol. I.

Table L-1. Fuel Cell System Failure Rate Estimate (Based on Allis Chalmers System) (Cont)

Part of Component Description	Quantity	λ	Total Quantity x λ
MRC (Moisture Recovery)			
H_2O Valve	1	3.000	
Transistor	1	.650	
Transducer	1	2.500	
Transducer	1	2.500	
Fittings		.600	
			6.750
WRS (Water Recovery)			
Inlet 3-Way	1	.240	
Pump 3-Way	1	2.600	
Pressure Relief	1	.800	
Gas/Stop/Reg.	1	1.000	
Check Valves	2	.300	
Pump Diaphragm	1	3.500	
Pressure Regulator	1	1.000	
Deionizer	1	.100	
Condenser	1	.350	
Fittings		2.400	
			12.300
EMCA (Electrical Monitoring and Control Assembly			
Cabinet	1	2.000	
H_2O Cavity	1	10.000	
Heating Cont.	1	2.000	
Purge Controller	1	15.000	
			29.000

APPENDIX M

CSM METEOROID FAILURE CRITERIA*

The Base Line CSM modes of failure due to meteoroids that affect the probability of crew survival and mission success are as follows: These failure modes imply that any meteoroid damage less than that described will not cause mission abort or crew loss.

COMMAND MODULE

Heat Shield Ablator - Any full depth penetration of the ablator through the outer face sheet of the substructure will allow gas to flow through the core during re-entry resulting in a catastrophic failure, crew loss, and mission failure. Maximum average ablator hole diameter is one-fourth inch.

Heat Shield Window Panes - Any penetration to the heat shield outer pane which results in any damage to the inner pane will cause the inner pane to fail on re-entry resulting in excessive cabin temperature, crew loss, and mission failure. If 10 percent of the total window viewing area is obscured by impacting meteoroids, the mission will be aborted.

RCS Engines - Any full depth penetration of any nozzle extension will allow hot exhaust gases to enter the outer equipment compartment resulting in a catastrophic fire, crew loss, and mission failure.

CSM Umbilical - Any full depth penetration of the inner shell will cause penetration of the wires, hardlines, or co-ax cables and will result in loss of power, crew loss, and mission failure.

SERVICE MODULE

SPS Oxidizer Tanks - Any twenty-five percent penetration of the wall thickness of any tank will result in an explosive decompression of that tank destroying a major part of the Service Module and resulting in crew loss and mission failure.

RCS Fuel Tanks - Same as SPS oxidizer tanks

RCS Helium Tanks - Same as SPS oxidizer tanks

*Based on NR SD data as reflected in Reference 2.7 of Vol. I.

SPS and RCS Plumbing - Any twenty-five percent penetration of the plumbing to any tank will result in an explosive decompression of that line, loss of propellant, crew loss, and mission failure.

EPS Cryogenic LO_2 Tanks - Any impact on the inner wall of a tank will cause a fracture and allow a complete loss of LO_2 from that tank. Failure of one tank will result in a mission abort. Failure of two tanks will result in failure of the EPS and crew loss.

EPS Cryogenic LH_2 Tanks - Any impact on the inner wall of a tank will cause a fracture and allow a complete loss of LH_2 from that tank. Failure of one tank will result in a mission abort. Failure of two tanks will result in failure of the EPS and crew loss.

EPS Cryogenic N_2 Tanks - Any impact on the inner wall of a tank will cause a fracture and allow a complete loss of N_2 from the tank, crew loss and mission failure.

SPS Engine - Any full depth penetration of the engine injector will result in inter-propellant mixing, catastrophic engine failure, crew loss, and mission failure. Any full depth penetration of the engine ablative thrust chamber either in the wall or throat section will create an untenable situation which will result in ultimate failure of the engine due to continual erosion in the area of penetration, crew loss, and mission failure. Any penetration of the engine nozzle extension resulting in a hole of greater than 3/8 inch in diameter will allow a burn-through of the nozzle extension resulting in failure of the SPS, crew loss, and mission failure.

RCS Engines - Any full depth penetration of an engine combustion chamber or nozzle will result in failure of that engine. Crew loss and mission failure results when both roll engine in the opposite quadrant and one roll engine in the opposite quadrant are not operable, or when two pitch-up engines in opposite quadrants and two pitch-down engines in opposite quadrants are not operable.

ECS Radiators - Any full depth penetration of an ECS radiator circuit will allow the coolant to escape resulting in failure of that circuit. Failure of the primary results in a mission abort. Failure of both the primary and the isolated circuits result in failure of the ECS, crew loss, and mission failure.

EPS Radiators and Fuel Cells - Any full depth penetration of a fuel cell will allow the electrolyte to escape resulting in failure of that fuel cell. Any full depth penetration of an EPS radiator circuit will allow the coolant to escape resulting in failure of that circuit. Since each fuel cell is connected to a radiator circuit, puncture of two fuel cells, two radiator circuits, or one fuel cell and one radiator circuit not connected to the punctured fuel cell will result in a mission abort. Puncture of all fuel cells, all radiator circuits, three fuel cells and the remaining radiator circuit, two fuel cells, and the remaining radiator circuits, or one fuel cell and the radiator circuits not connected to the punctured fuel cell will result in failure of the EPS and crew loss.

DSIF Antenna - Any full depth penetration of the antenna support boom will sever the cable inside and result in loss of communication and a mission abort. The antenna is not essential for crew survival.

Electrical Wiring System - Any full depth penetration of the insulation of any electrical wire will result in the severing of that wire. This will result in failure of that system, crew loss, and mission failure.

APPENDIX N

SNAP-27 RADIOISOTOPE THERMOELECTRIC GENERATOR (RTG) FOR ELOR MISSION APPLICATION*

SNAP-27 SYSTEM DESCRIPTION

The SNAP-27 generator is the electrical power source for the ALSEP system being developed for the NASA Manned Spacecraft Center. The ALSEP is a series of instruments and supporting subsystems that will be deployed on the lunar surface by the astronaut and will transmit lunar environmental information for a period of at least one year.

In the Apollo mission, the de-fueled generator is transported to the lunar surface in the scientific equipment bay of the LM. During transit, the fuel capsule is placed in a special container attached to the exterior of the LM Descent Stage. After landing, the astronaut removes the generator from the equipment bay (as part of the ALSEP package) and places it on the lunar surface. He then extracts the fuel capsule from its container and inserts it into the generator. A special flight handling tool is provided for this purpose. The over-all ALSEP is then carried to its final location and deployed for operation.

The SNAP-27 system itself is being developed by the General Electric Company for the Atomic Energy Commission. The power supply consists of two major components, the generator assembly of Figure N-1, and the radioisotope fuel capsule assembly of Figure N-2. The thermal energy provided by the Pu-238 radioisotope fuel contained in the fuel capsule is partially converted to electrical power in the generator assembly. The remainder of the energy is rejected by the fin heat rejection system. To simplify the fueling operation, energy is transferred from the fuel capsule assembly to the generator assembly by radiation. Characteristics of the SNAP-27 generator, as developed for the ALSEP mission, are presented in Table N-1. Structurally, the generator assembly is designed to withstand the dynamic environments associated with the Apollo mission. The structure is also designed to allow repeated insertion and withdrawal of the fuel capsule without damage to the thermopile or the structure.

The fuel capsule assembly is a cylinder 2.5 inches in diameter by 15.6 inches long. It is attached to an end plate containing the latching mechanism by which it is attached to the generator assembly.

*Provided by General Electric Co. through Reference 4.18.

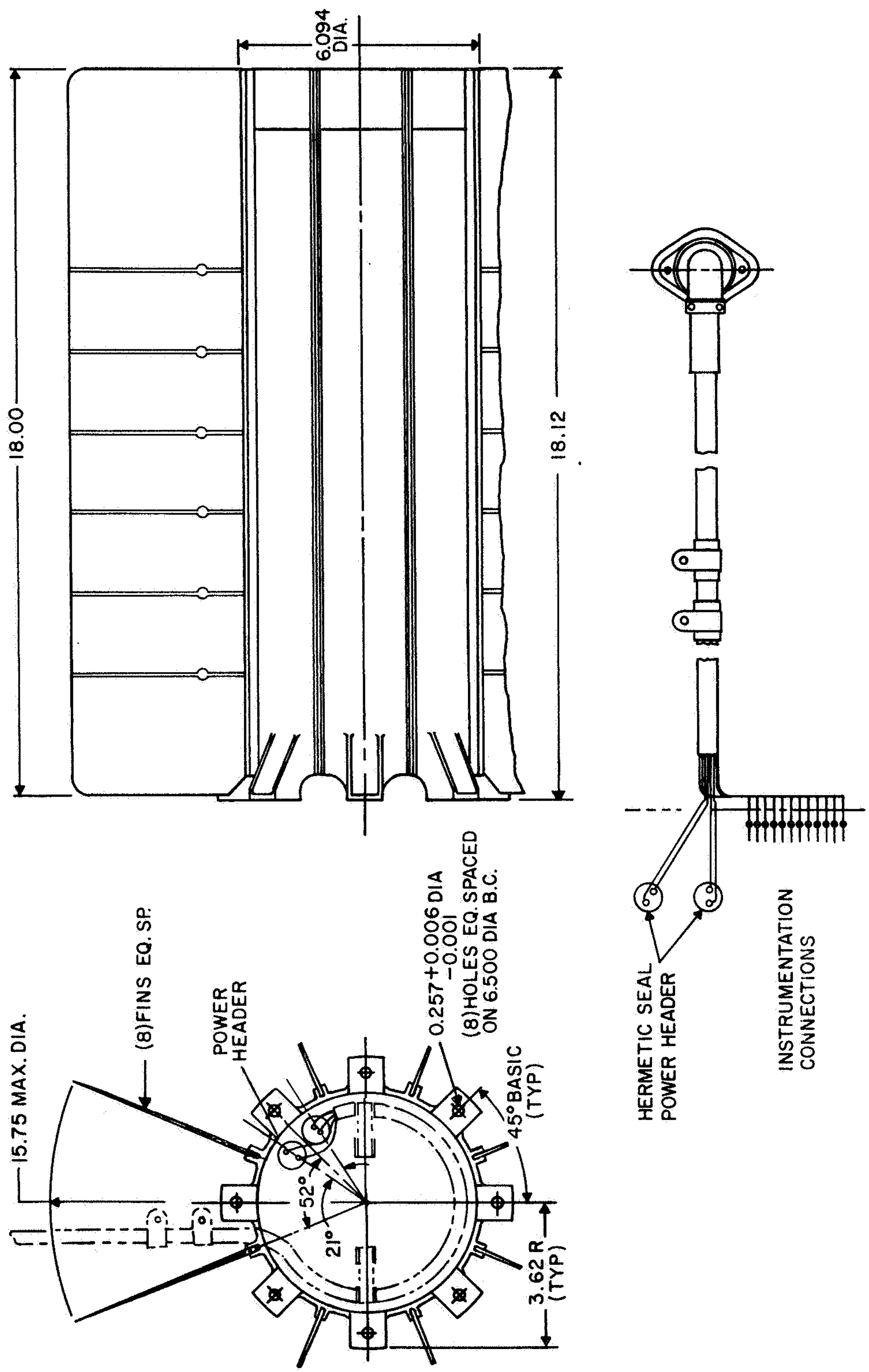


Figure N-1. Generator Assembly Configuration

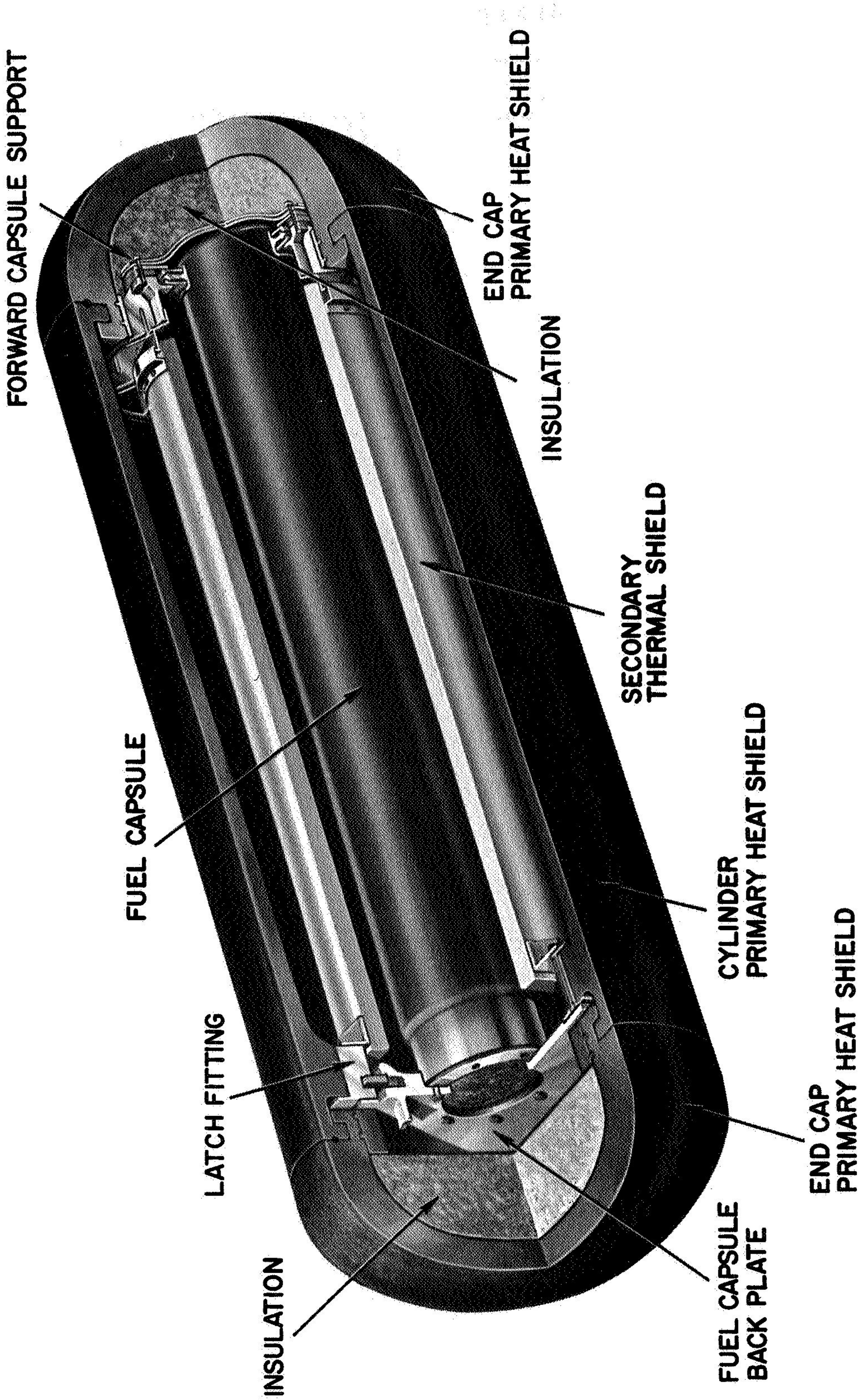


Figure N-2. Graphite LM Fuel Cask

Table N-1. SNAP-27 Radioisotope Thermoelectric Generator Specifications

Item	Characteristic	
Mission Application	Power Apollo Lunar Surface Experiment Package (ALSEP).	
Conversion Concept	PU-238 Fueled Thermodelectric System Using Lead-Telluride Alloy Thermocouples in Two Series Parallel Strings.	
Design Life	One Year Lunar Operation Preceded by Two Years Earth Storage.	
Generator Performance		
Output Power		
Specified (watts)	65 (BOM*)	63.5 (EOM)*Min.
Measured (watts)	73.3 (BOM)	68.5 (EOM)**
Output Voltage (nominal)		16 volts DC
Current (nominal)		4 amps
Over-all Efficiency (nominal)		4.75%
Average Hot Junction Temperature		1075° F (580° C)
Average Cold Junction Temperature		525° F (271° C)
Fuel Capsule Thermal Output (nominal)		1450 watts
Mechanical Characteristics		
Over-all Diameter Over Fins		15.7 inches
Over-all Length		18.1 inches
Number of Fins		8
Fin Radial Length		5.0 inches
Fin Axial Length		18.0 inches
Weight		
Generator Assembly (includes Cable, Connector, and Instrumentation)		28.2 pounds
Radioisotope Fuel Capsule Assembly		14.5 pounds
Fueled Generator		42.7 pounds
*Beginning (or End)-of-Mission **1485 Watt Thermal Input, 12,113 hours		

The graphite LM fuel cask (GLFC) supports the fuel capsule assembly during the transit phase and also provides re-entry protection in the event of an orbital or superorbital abort. Graphite is the principal heat shield material, of which the secondary thermal shield is beryllium. The GLFC is 23 inches long by 8 inches in diameter and weighs 25.75 pounds.

The final piece of flight hardware is the handling tool which transfers the fuel capsule assembly from the GLFC to the generator assembly.

The current program status is:

Generator Assembly - qualification is complete. All flight units (5) are assembled and in storage pending shipment to NASA.

Fuel Capsule - qualification is complete. All fuel capsule assemblies are fueled and are currently being prepared for delivery.

GLFC - development is complete. Combined systems qualification with Bendix support hardware is currently in process.

Support equipment - all fabrication and qualification is complete.

OPERATIONAL PROCEDURE RECOMMENDATIONS

Based on the Operational Choices presented in paragraphs 4.4.1 of Volume I, the resultant operational sequence recommended for the LM is as follows:

Prelaunch

1. The generator assembly and the GLFC is mated with the LM prior to the time the latter enters final system test. This will allow checkout of the generator and the storage power system during such tests. Either the nuclear fuel capsule assembly or the electric fuel capsule simulator may be used to heat the generator for these tests.

2. Late in the countdown, the fuel capsule assembly is placed in the GLFC. For the present mission, this occurs at approximately T-16 hours.

Launch and Transit

No operations required as the system is passive.

System Activation

1. If required, the generator is deployed to its final operating position.

2. The flight handling tool is assembled. This involves attaching the jaws to the extension handle.

3. The GLFC is rotated to the fuel capsule transfer position (basically horizontal).

4. Using the flight handling tool, the end cap of the GLFC is removed.

5. Again using the flight handling tool, the fuel capsule assembly is unlatched and removed from the GLFC.

6. The fuel capsule assembly is inserted and latched in; full power is available in approximately 30 minutes.

System Operation

No operations other than monitoring are required during operation. No maintenance or repair of the power supply is required or possible.

Time for the fueling operation is expected to be less than ten minutes.

During operation, the generator will deliver a constant power to some type of regulator/protective circuit. Since the generator has no overload capability, any power demand in excess of its power delivery must come from batteries.

POWER SUPPLY RELIABILITY

High reliability represented a key goal in the development of the SNAP-27 generator. This was achieved by continuing reliability assessments throughout the program, by a comprehensive quality assurance program applied to all hardware, and extension testing of both modules and generators. Reliability assessments included a detailed failure mode and effects analysis to establish and eliminate if possible potential weak points of the system. Examples of design features incorporated as a result of this effort include the series-parallel electrical network, the all welded hermetic seal and the dual positive and negative leads from the generator.

The thermopile design was verified using special test modules and prototype generators:

- A 10-couple design with the elements mounted in a flat plate array. These modules were primarily used for material stability tests. A total of 19 such modules were built, 15 of which remain on test having already accumulated 12,000 - 18,000 hours of operation.

- A 104-couple design with the elements mounted in the same manner as the generator. All radial dimensions of these modules duplicated the generator. In essence, the module is a shortened SNAP-27. The modules were used for efficiency, dynamic loading, and material stability studies. Eleven modules were tested, ten of which remain on test having accumulated 6,000 to 11,500 hours of operation.

In addition to the modules, three generators have been tested in a simulated lunar environment. Operating times are 11,513 hours, 7,741 hours and 7,070 hours (as of 6/15/68). Each generator has operated as predicted prior to the tests.

Based upon this testing, which gives over 20 million couple hours with no apparent failures, the reliability of the generator for the ELOR 90-day mission has been demonstrated to be greater than 0.9999 at 50% confidence; in actuality it is probably much greater, approaching 1.0. This includes allowance for electrical failure through short or open circuit, failure of the hermetic seal, or degradation at a greater than predicted rate which would result in a less than predicted minimum power capability after 90 days.

GROWTH CAPABILITY

Although the current SNAP-27 power supply is expected to meet the ELOR mission power requirement; however, it is possible that other power levels may be required following more detailed analyses. Growth characteristics are therefore worth considering, particularly for application to the CSM requirement.

The basic design is compatible with mission power requirements between 20 to 140 watts (e). These requirements are met by simply shortening or lengthening the current design, accomplished by the addition or deletion of the necessary number of the circumferential rows of the thermoelectric elements. The radial dimensions of the thermopile, as well as the internal geometry and couple hardware, remain the same as the present SNAP-27 unit. This ability to meet various power requirements without altering the internal thermopile materials and geometry permits application of the extensive test data to designs directed toward other power levels. Therefore, the need for and the extensive costs related to a new development program, which would otherwise be required to assure long life performance and reliability, is eliminated.

A key parameter in generator design is the hot junction temperature. This in turn requires consideration of the mission life requirements since power output degradation of the unit increases with increasing hot junction temperature. Thus, initial increases in thermodynamic efficiency, related to the higher hot junction temperature can be negated by the associated higher thermoelement degradation rates. This results in end-of-mission (EOM) power levels appreciably below those achieved at BOM and below those achieved by starting at a lower temperature. The range of hot junction temperatures which can be considered at present is about 100°F. Long term thermoelement test data indicates that the hot junction temperature may be as high as 1075°F for one year missions, but it must be lowered to about 1025°F if a five-year mission is to be considered. Still higher hot junction temperatures might be used for a 90-day mission, however, this is still under study. For this study, the maximum hot junction temperature considered is 1075°F, resulting in a generator optimization for a one-year life.

Performance characteristics of SNAP-27 type RTG systems are presented in Figures N-3 and N-4 as a function of power level from 20 watts (e) to 150 watts (e).

These data predict RTG characteristics with an accuracy of better than 90%, assuming that:

1. The thermopile is identical to that employed in the current SNAP-27 generator, except for length.

2. All electrical power levels are power at the end of a five-year mission.

3. The generator is launched with the fuel capsule installed. Integral fuel capsule re-entry protection is provided by a high density graphite heat shield on the outside of the fuel capsule. The graphite weight is included in the weight.

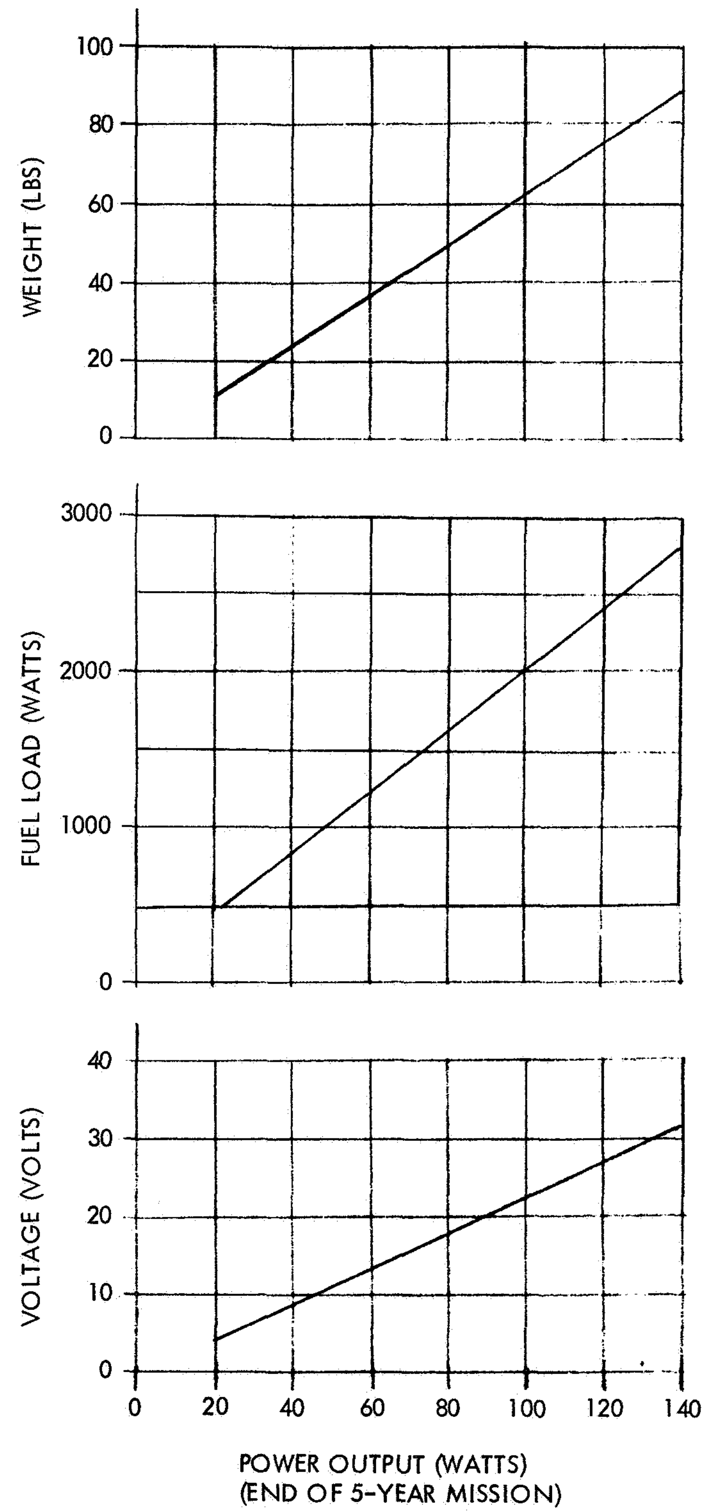


Figure N-3. SNAP-27 RTG System Performance

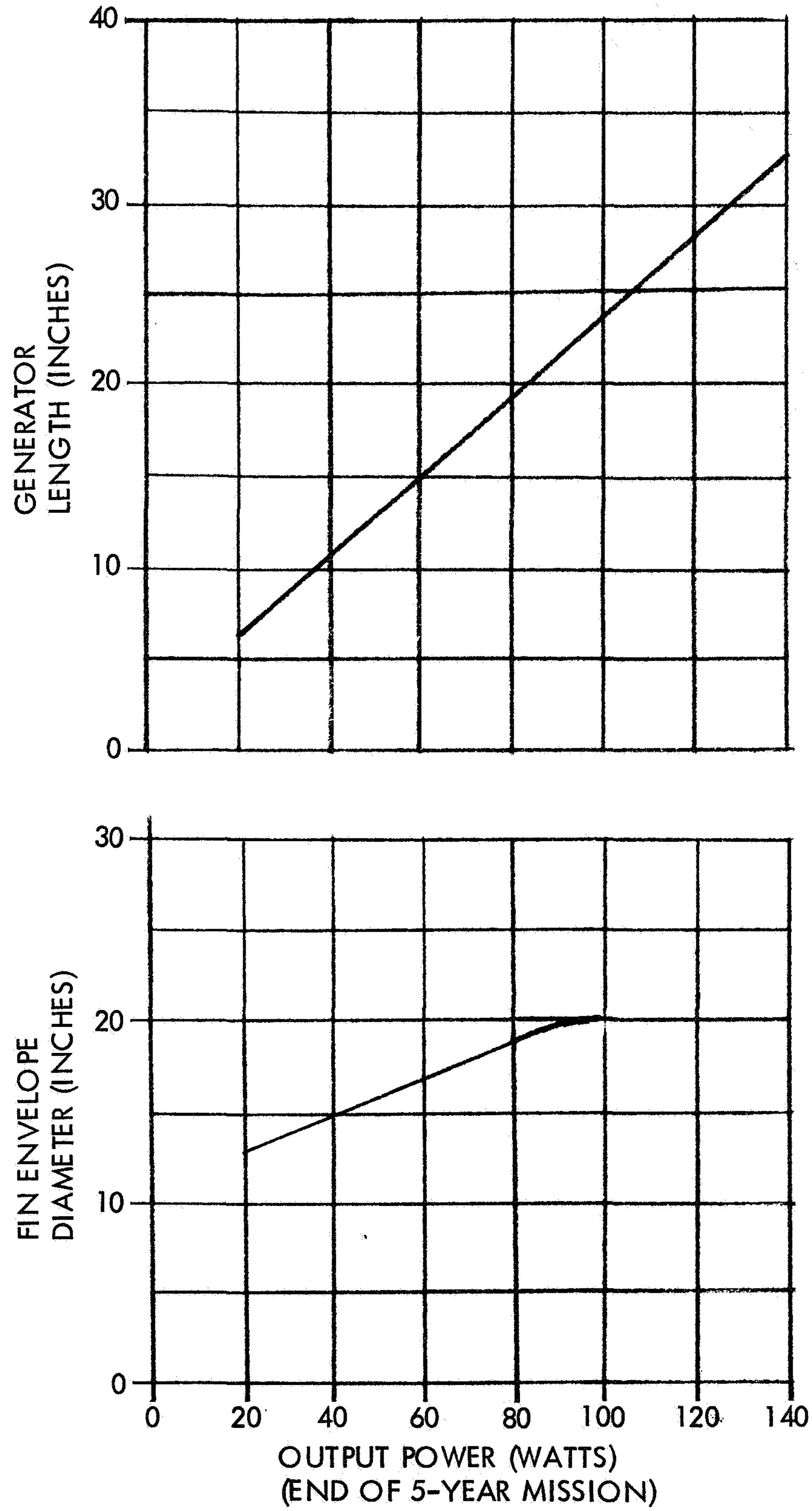


Figure N-4. SNAP-27 RTG System Geometry

RTG data are readily corrected for the shorter one-year mission time using the following relations:

$P_1 = 1.28\ P_5$	P denotes electric power, watts
$Q_1 = 1.09\ Q_5$	Q denotes fuel inventory, watts
$W_1 = 1.03\ W_5$	W denotes weight, pounds
$V_1 = 1.09\ V_5$	V denotes voltage
	1 denotes one-year mission
	5 denotes five-year mission.

HEAT TRANSFER TECHNIQUES

There are four alternatives for coupling heat energy into the LM and/or away from the SNAP-27. These are illustrated in Figure 4 of Volume I. They operate as follows:

1. Radiation coupled unit boiler - With the generator mounted vertically, this is the simplest configuration. In addition, it provides thermal shielding of the LM during lunar day, preventing the addition of unwanted heat at this time. Its principal disadvantage is the radial thermal gradient established in the generator, the magnitude of which will depend upon the amount of blockage, and will be worse during lunar day when heat is not being removed from the boiler. Such gradients can reduce the power output of the generator as well as cause additional operating stresses. However, calculations indicate that for nominal blockage, the gradients will not be excessive.

2. Radiation coupled segmented boiler - This approach can be taken instead of the unit boiler if the gradient proves to be unacceptable. It can also be used if redundant loops are desired in the boiler system. Disadvantages include more complex fluid/mechanical design and lack of thermal shielding for LM during lunar day.

3. Radiation coupled wrap around boiler - This is a desired configuration if the generator is mounted horizontally. It eliminates the more severe radial gradient, and if it is not too long, very little axial gradient results. If a large amount of heat is required, however, the blockage can be excessive, leading to larger weight penalties than the unit boiler.

4. Conduction coupled boiler - The principal advantage is lack of thermal blockage of the generator, hence no fin weight penalty. Drawbacks include the more complex mounting arrangements for the generator and the more complex mechanical boiler design, (since it must now carry the launch loads of the generator). By extending the boiler under the fins, this arrangement may also take advantage of radiation to increase the heat input.

Radiant heat input to the boiler is shown in Figure N-5 for various boiler surface temperatures based upon the simplified definition of thermal blockage shown in Figure N-6. In addition, the additional fin weight necessary to maintain the generator temperature is shown by the dashed line on Figures N-7 and N-8 for vertical and horizontal mounting arrangements respectively. The effect of permitting higher fin base temperatures is also illustrated on the curves. In deriving these results, it was assumed that during lunar day the boiler would be inoperative and would therefore act as an adiabatic wall. Depending upon the specific geometric relationships, this is somewhat conservative, and the results shown represent upper limits.

It can be seen that up to 300 watts may be transferred without excessive weight penalties provided the fin base temperature is allowed to rise to 550° F during lunar day operation.

Conductive heat input to a conduction boiler can be controlled by shims at the mounting interface. While this system does cause an axial gradient, preliminary calculations indicate that 100 to 200 watts may be transferred in this manner without problems. In addition, if the boiler is extended to the ends of the fins, approximately 100 watts additional will be transferred by radiation. Therefore, this system capability is similar to the pure radiation approaches.

If more than 300 watts is desired, a combination system using a conduction coupled base boiler and a wrap-around boiler is recommended.

Undesired heat input to either vehicle can be minimized by use of multifoil insulation. Calculations indicate that this heat input can be limited to less than 30 watts, even assuming that the generator is mounted so that half of it is inside the spacecraft. With the recommended mounting approach outlined below, the expected values will be significantly less than this figure.

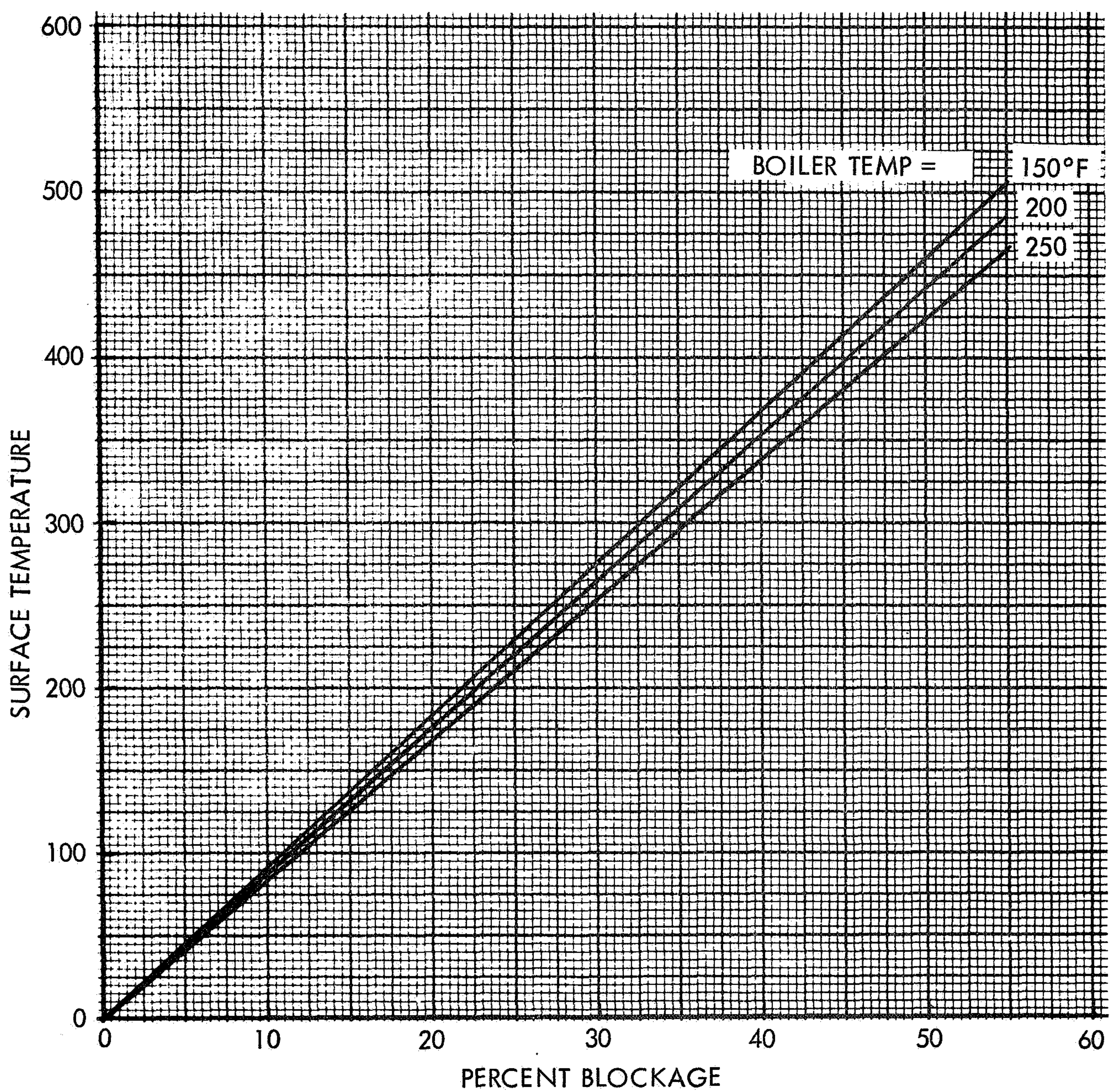


Figure N-5. Heat Into Boiler as a Function of Blockage

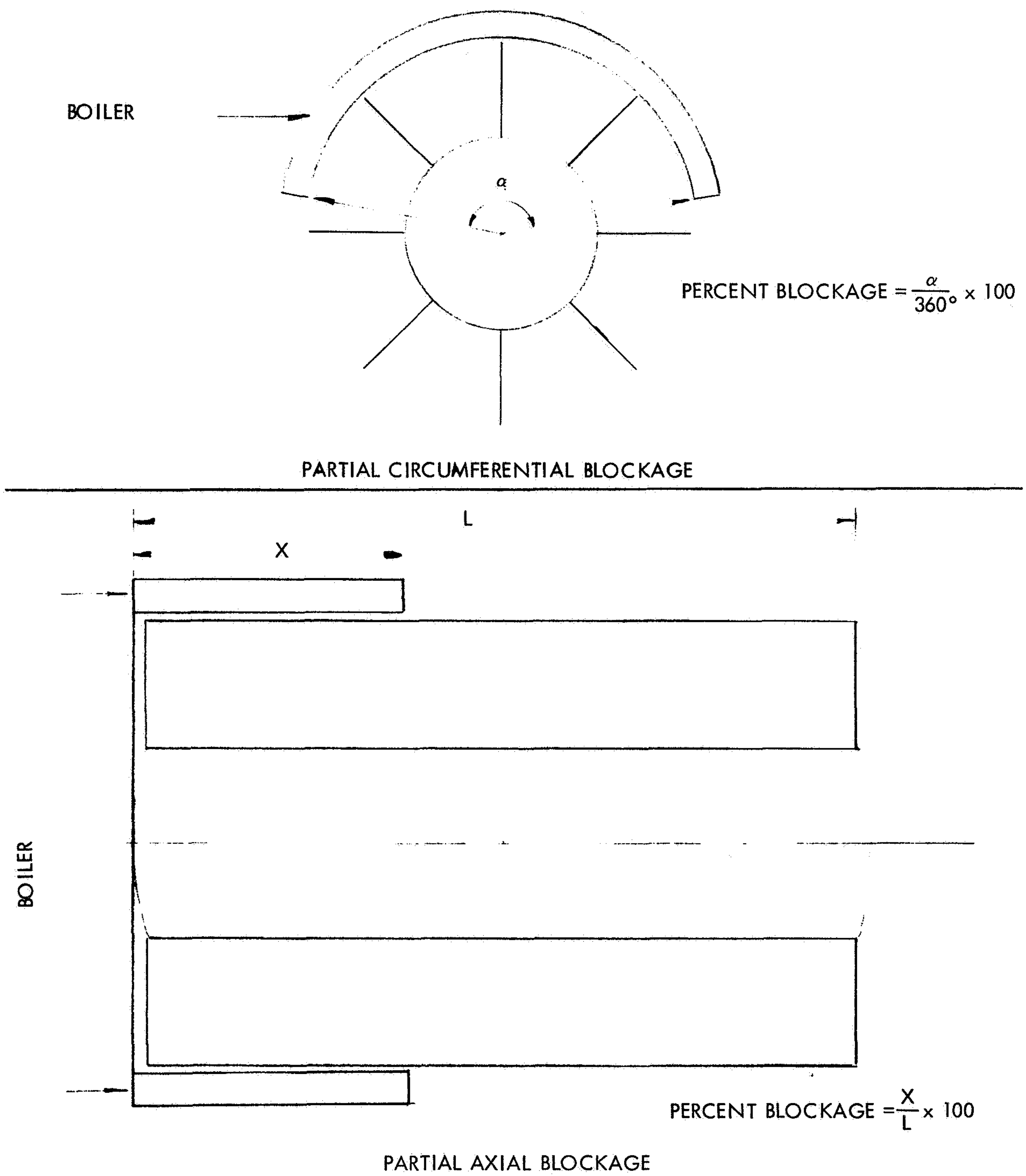


Figure N-6. Boiler Configuration and Blockage Definition

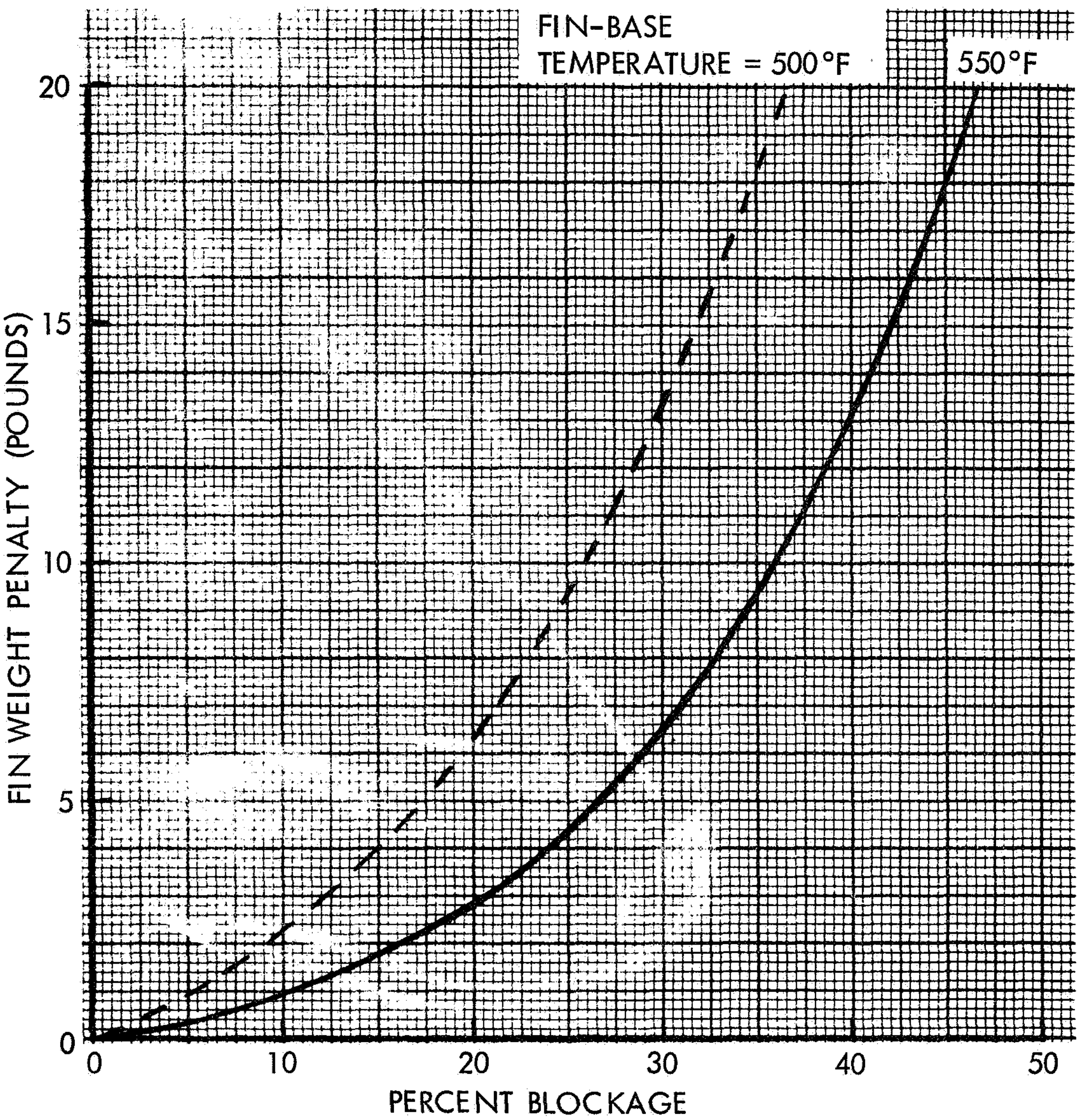


Figure N-7. Fin-Weight Penalty for Vertical Generator as a Function of Blockage

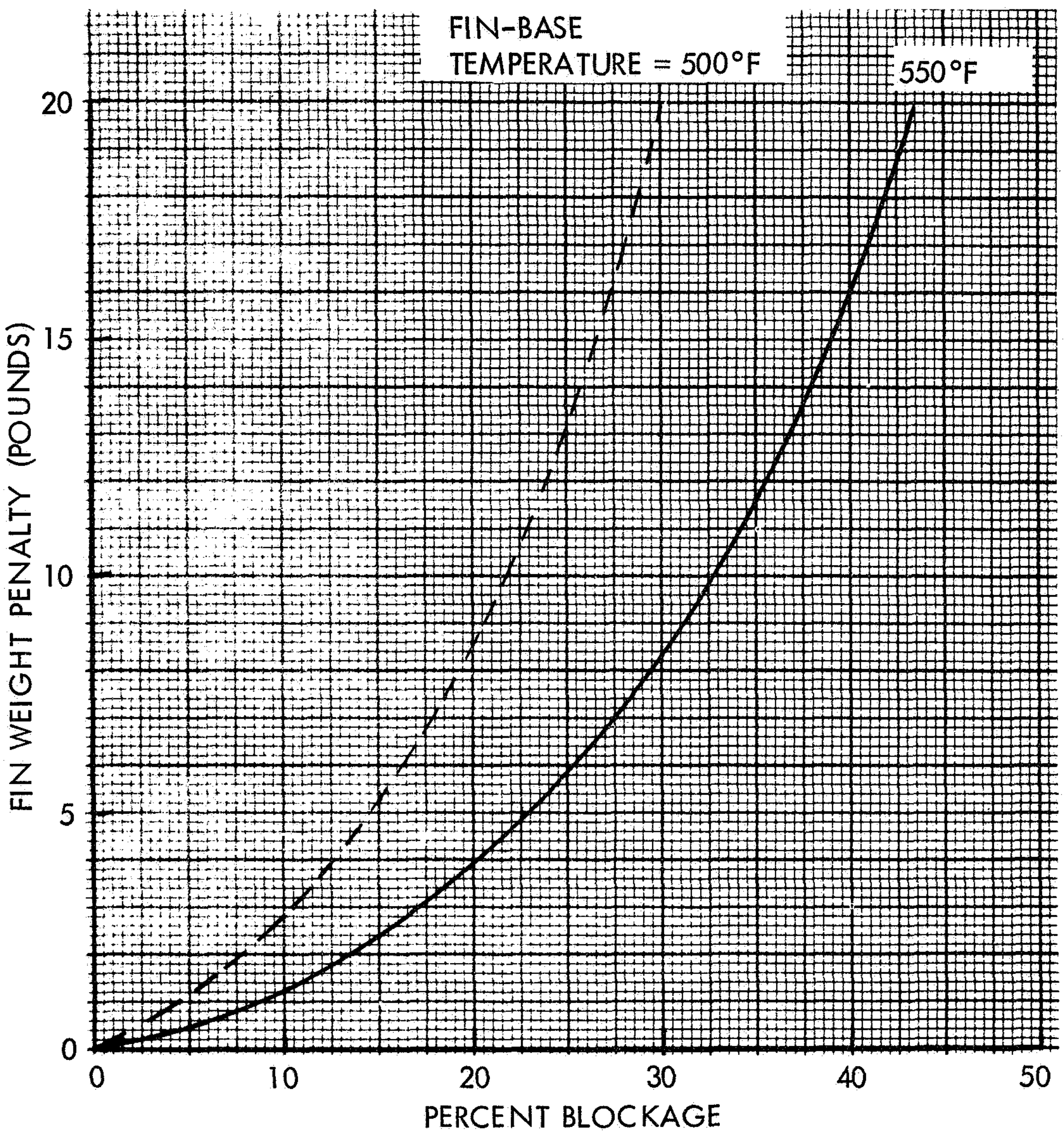


Figure N-8. Fin-Weight Penalty for Horizontal Generator as a Function of Blockage

Thermal characterization of the recommended LM RTG is given in the following:

Radiator surface emittance (ϵ) is considered to be 0.85 as a minimum with an assumed absorptance (α) of 1.00. For preliminary integration studies, equivalent cylinders with the following characteristics may be used if the α/ϵ is considered to be 1.0:

	Dia. (in)	Length (in)	T°F
Lunar Day	15.7	18.1	433
Lunar Night	15.7	18.1	359

Following insertion of the fuel on the lunar surface, the time to thermal equilibrium will be approximately 1.5 hours. Full power may occur earlier depending on initial temperatures, sink temperatures and electrical load. Surface temperature of the unit will vary from 510°F at the face of the cylinder during lunar day to 275°F at the tip of the fin during lunar nights with temperature change at any point approximately 50° from lunar day to lunar night conditions.

INSTALLATION CONSIDERATIONS

Mechanical

Since the generator is shipped in the scientific equipment bay, it is best operated in the same location so that transport on the lunar surface is not required. During operation, the fins must be exposed to allow heat rejection, and the mounting must take this into account.

The generator could be launched in its final operation position; however, this has certain disadvantages. An RCS nozzle is located above the scientific equipment bay, and the plume would hit the generator with unknown effects on thermal coatings, etc. Rocks and dust could impinge on the fins during the lunar landing. In addition, the need to clear the SLA during LM withdrawal limits the amount of the generator which extends beyond the compartment. Therefore, the recommended approach consists of mounting the generator inside the compartment and "racking" it out prior to operation.

Figure N-9 illustrates this approach for both a vertical and a horizontal mounting. The former is similar to the current ALSEP mount and may be preferred for this reason. However, the horizontal mount has also been analyzed in detail, and no apparent problems have been uncovered.

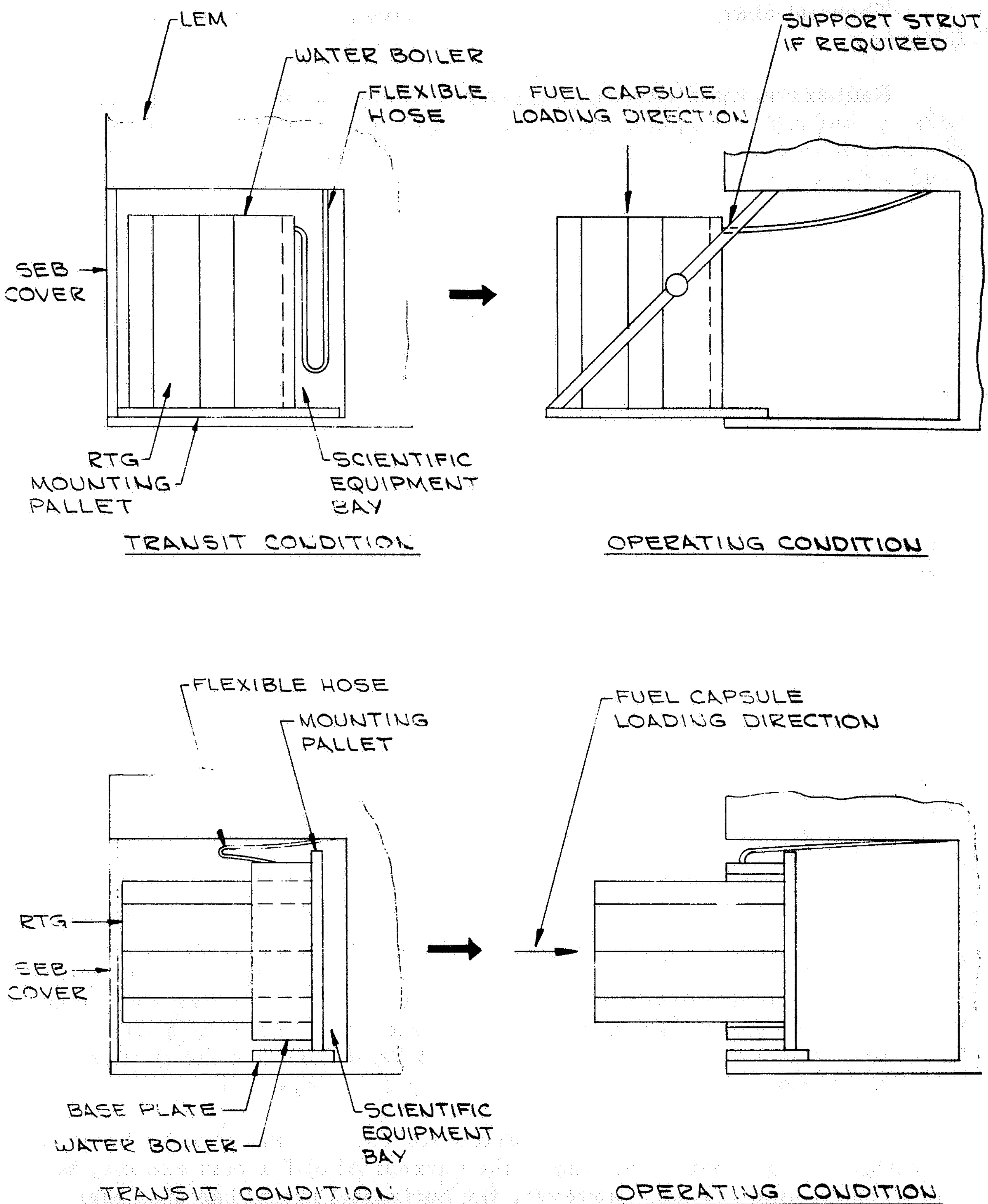


Figure N-9. Electrical Power Generating Assembly Mounting Approaches, LM Vehicle

One aspect of the selection between the two approaches involves the ease of fueling. The horizontal system is better in this regard since it is the same orientation as the current GLFC. However, detailed studies will be required including the effect of inclination of the LM on the lunar surface.

The mechanical interface of the generator consists solely of the eight mounting lugs and the single Bendix type AN connector.

Electrical

Typical generator electrical characteristics are defined by the performance map shown in Figure N-10. Design output voltage is 16 ± 0.5 vdc. At the nominal voltage, current varies from 4.6 to 4.3 amperes, depending on the exact fuel loading.

A thermoelectric generator is essentially a constant power device with no overload capability. In addition, if the power output is reduced, Peltier cooling is reduced and the hot junction temperature increases. This leads to accelerated degradation. Therefore at all times during the operation life, and during operational testing, the generator must be operated in a "loaded" condition. This can be accomplished by limiting the voltage across the output terminals of the generator to no more than 16.5 volts using same form of voltage regulator. In essence, the voltage regulator then provides the additional function of a protective circuit.

Various types of RTG regulators are possible but the shunt dissipative type is usually selected for the following reasons:

1. This eliminates problems between the ripple on the dc bus and the inverters.

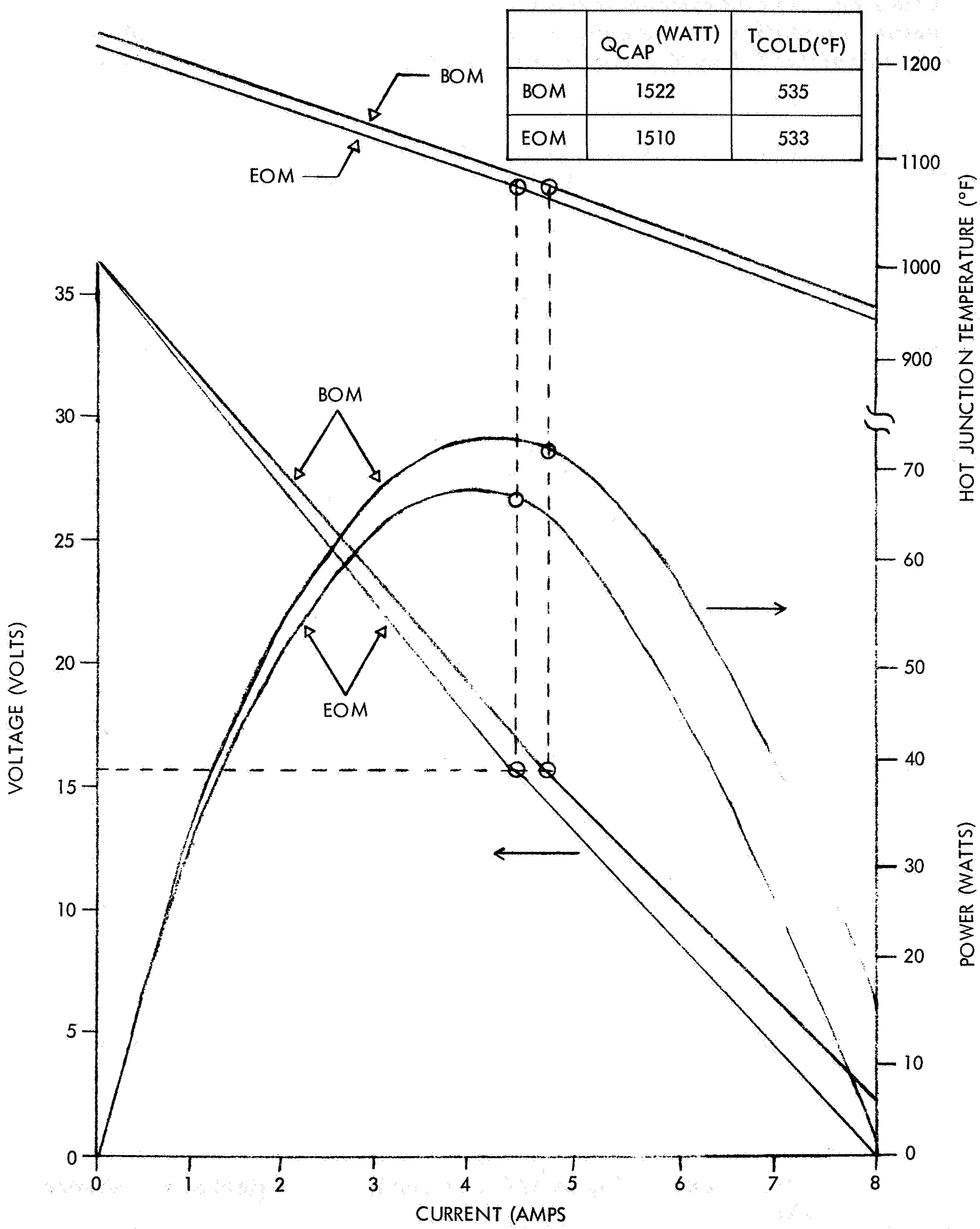
2. Circuits for the shunt regulators are simpler; no multivibrator required.

3. Filters are not required.

4. Inverter input voltage may rise during low load conditions with switching regulator.

5. Electromagnetic interference is reduced.

6. Transistor power rating is 1/4 that required if external resistance is not used.



	Q_{CAP} (WATT)	T_{COLD} (°F)
BOM	1522	535
EOM	1510	533

Figure N-10. Typical SNAP-27 Performance Characteristics (Calculated) for Lunar Day and Maximum Fuel Load

Magnetic Interface

Because of its intended use as the power supply for scientific instrumentation, special precautions have been taken in the design and construction of the SNAP-27 generator to minimize magnetic effects. These include:

1. Incorporation of degaussing loops in the generator. These diple degaussing loops (buckling coils) are incorporated at each end of the generator to limit the current induced magnetic field. Current flow in these coils is opposite to the circumferential current flow through the ends of the thermopile.

2. Elimination of all ferromagnetic material except the hot shoes and hot buttons used in the thermopile.

3. Deperming of the hot shoes and hot buttons prior to generator assembly.

4. Final deperming of the entire generator assembly before shipment to user.

An analytical and experimental program is being conducted to establish the final magnetic field values for the assembly.

Nuclear Radiation Characteristics

The plutonium-238 fuel decays by the emission of alpha particles that are either trapped within the fuel itself or contained by the capsule. With acquisition of electrons, these particles become helium atoms and cause a gradual pressure build-up in the capsule. Low level neutron and gamma photon currents are also emitted and penetrate the surrounding materials. However, the levels are sufficiently low that no special shielding is needed to install the fuel capsule at the launch site or on the lunar surface.

The neutrons are produced from spontaneous fission - directly and by neutron multiplication - and by alpha-neutron reactions with light elements. The gamma activity arises from the prompt fission gammas, from the decay of fission products, and from alpha-neutron reactions. Secondary radiation fields resulting from interactions are negligible.

Based upon a measured meutron production rate, the neutron flux at one meter from the generator is 640 neutrons/cm^2 sec. For distances beyond one meter, the inverse square law is directly applicable. The anticipated neutron flux spectrum "peaks" at 2.2 mev. and relative values at other energies are given in Table N-2.

Table N-2. SNAP-27 Nuclear Radiation Characteristics

Neutron Flux (Normalized)

Energy (mev)	E/ (2.2 mev)	Energy (mev)	E/ (2.2 mev)
0.05	0.085	2.4	0.99
0.4	0.12	2.8	0.86
0.8	0.22	3.2	0.55
1.2	0.52	3.6	0.27
1.6	0.82	4.0	0.14
2.0	0.98	4.4	0.085
2.2	1.00	4.8	0.062

Gamma Spectra (Photons/cm^2-sec)

Energy (mev)	1 Day Old PU-238	Energy (mev)
0.04 - 0.5	8.6 (--2)*	7.9 (--2)
0.50 - 1.0	1.0 (3)	1.5 (3)
1.00 - 2.0	7.3 (1)	1.6 (2)
2.00 - 3.0	5.5 (1)	1.5 (3)
3.00 - 5.0	1.1 (1)	9.8 (0)
5.00 - 7.0	4.9 (0)	4.2 (0)

*(n) = 10^n

The spectrum and specific yield of gamma photons is influenced by the initial concentration of Pu-236, since a decay product - thallium-208 - produces a penetrating 2.6 mev gamma during decay. Tl-208 achieves a maximum concentration approximately 17 years after fuel production, and, therefore, the values are also time dependent. Typical gamma spectra at one meter from the generator are shown in the table. Allowance has been made for the fuel geometry and attenuation due to self-shielding effects of the fuel capsule and generator materials. Simple build-up factors have been used to allow for photon scattering.

If these data are used to obtain tissue dose rates, the following values result for a one meter separation distance:

Neutron	70 mrem/hour
gamma (1 day aging time)	2 mr/hour
gamma (5 year aging time)	8 mr/hour

APPENDIX O.

P&W FUEL CELL SYSTEM FOR THE QUIESCENT LM*

SYSTEM DESCRIPTION

The power requirements of the LM on the lunar surface can be met by a lightweight, low-temperature fuel cell power plant system of state-of-the-art design. A system consisting of three fuel cell modules, one operating for normal loads and two operating for extended peak loads with one module as a spare, is recommended. Short peaks are supplied by on-board batteries or the redundant module.

Fuel cell heat rejection requirements during the lunar day are easily met by using a water boiler or sublimator in addition to a small radiator. More than enough water is produced during the lunar night by the fuel cells to provide heat rejection capability during the lunar day, regardless of the radiator orientation.

The fuel cell power plant system concept proposed for the LM involves three modules. One is for normal power, the second for peaking, and the third for redundancy. They are cooled with a common coolant loop configuration. The individual heat and product water removal subsystems are self-regulating over the required range of power levels. Multiple modules with a common accessory section have been used in other P&WA power systems most notably the PC10, a low-temperature base-electrolyte power plant.

The product water is separated and stored in the accumulator until it is required for heat rejection purposes in the water boiler or sublimator.

The coolant regenerator is included in the heat rejection subsystem to prevent excess heat rejection during low power levels and low sink temperatures. The regenerator is bypassed during high power and high sink temperature periods.

Reactant preheaters are included to bring inlet H_2 and O_2 reactant temperatures from cryogenic to operating levels.

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* Much of the data used herein was provided by Pratt and Whitney.

The water boiler, or sublimator, is automatically controlled to provide auxiliary heat rejection as required during the lunar day to maintain the proper module coolant inlet temperature. The water boiler is bypassed during low power levels.

Two fuel cell radiator configurations were considered. Either configuration will meet the mission heat rejection requirement during the lunar day/night cycle.

Radiator Mounted Horizontal on Top of Vehicle and Shielded from Lunar Surface

For this case, the external heat input to the radiator is equivalent to that of a vehicle in space because there is no incident radiation from the lunar surface. The incident radiation will vary from a maximum of 440 Btu/hr-ft^2 when the sun is directly overhead to approximately zero during the lunar night. Assuming a radiator coating emissivity of .85 and an absorption of .20, the maximum equivalent sink temperature is 35°F. For a 50 watt fuel cell design power, a 1.5 ft^2 radiator is required with a 160°F average radiator temperature required for 100-watt fuel cell output is reduced to 100. F. No evaporation of product water is required during the mission.

Vertical Radiator Mounted Circumferentially Around Vehicle

During the lunar night the equivalent sink temperature is -276°F. If the radiator temperature is allowed to decrease to -35°F (minimum CSM radiator temperature), at 100 watts, a 5 ft^2 radiator can be used. During the lunar day this radiator will reject the necessary heat up to a maximum sink temperature of 135°F, or for approximately 60 percent of the lunar day. During the 40 percent period of maximum sink temperature, a maximum of 21 pounds of water will be evaporated to maintain the fuel system in equilibrium; this is less than 40 percent of the water produced per day/night cycle. For the first day, assuming a landing at the time of maximum sink temperature, approximately 12 pounds of additional water will be required for supplemental cooling. After the first day, excess water will build up until at the end of 90 days, approximately 110 pounds will be available.

WEIGHT

Estimated system weights are provided in Table O-1.

Table O-1. Estimated System Weight

System	Weight (pounds)
Three Fuel Cell Modules and Required Accessory Components	140
Spare Radiator (5 ft^2)	10
Reactants (including 1 percent outage, 1 percent purge)	180
Reactant Tankage	80
Water (to be evaporated during 1st lunar day)	10
Total	420

The load profile of Table O-2 was utilized in calculating the reactant weights of Table O-1.

Table O-2. LM Dormant Stay Load Profile

Total Net Power (watts)	Time (hours)
50	336
100	360
50	336
100	336
50	336
100	336
50	90
Total Reactants - 180 pounds	

The reactant weights for the initial landing and pre-launch high power periods were not included due to the uncertainty of the duration of these loads. The reactant consumption at 1.0 kw (2 operating) will be approximately 0.87 pound per hour.

PERFORMANCE

The system performance as a function of power output for both one module and two modules operating is shown in Figures O-1, O-2, and O-3.

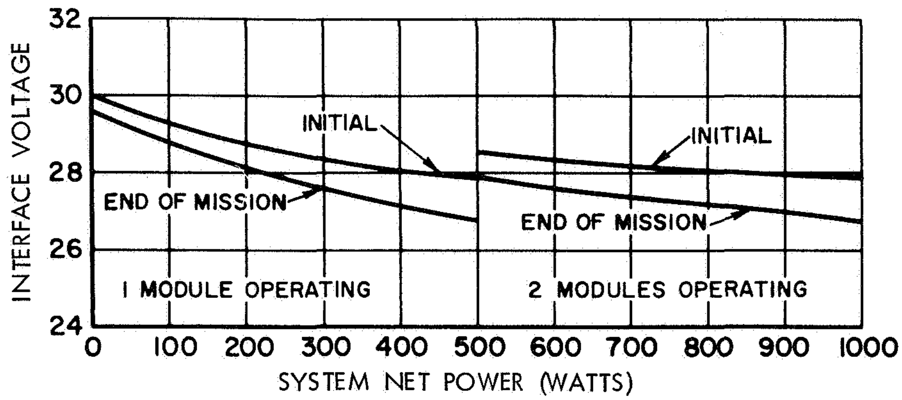

Figure O-1. Interface Voltage for LM Fuel Cell System

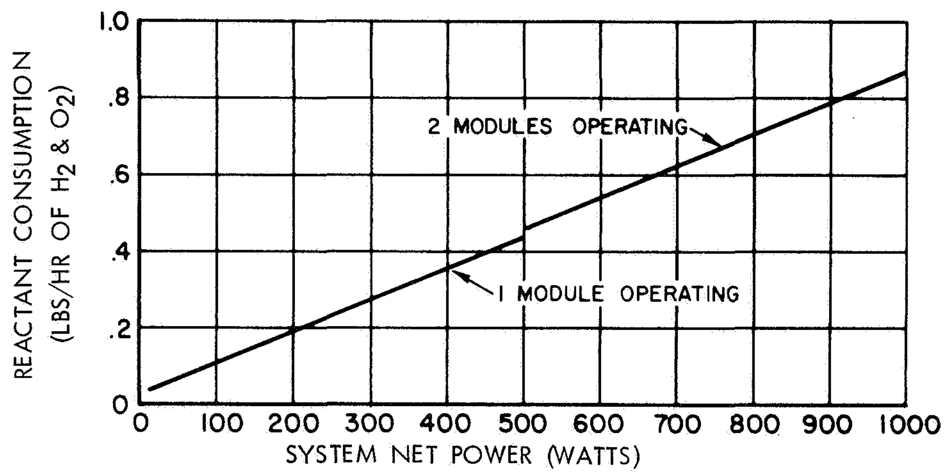

Figure O-2. Nominal Reactant Consumption for LM Fuel Cell System

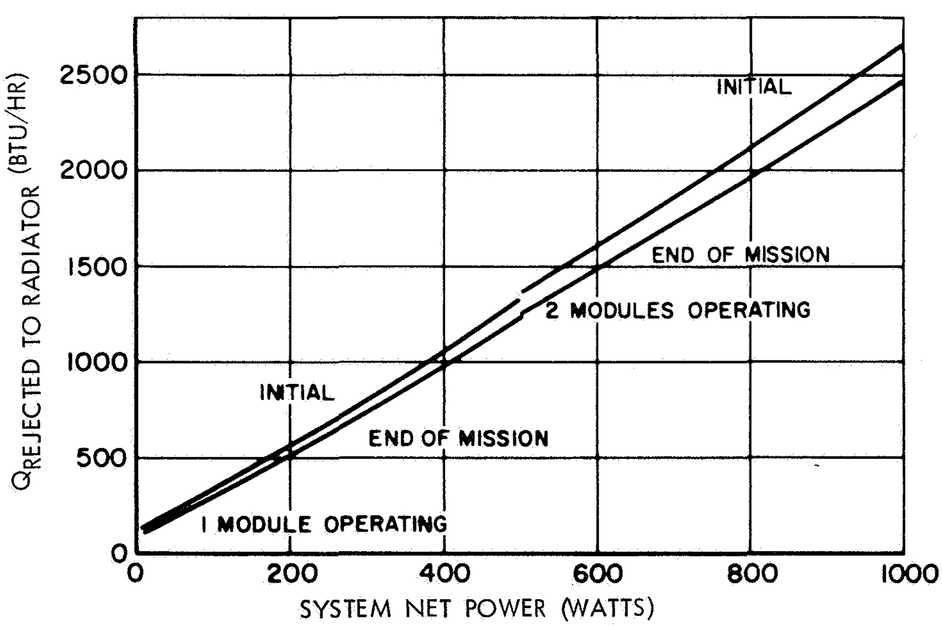

Figure O-3. Heat Rejection to Radiator for LM Fuel Cell System

APPENDIX P

ALLIS CHALMERS FUEL CELL SYSTEM FOR THE QUIESENT LM*

Allis Chalmers proposes a fuel cell module which appears ideally suited for the ELOR Lunar Module application. The fuel cell module is rated at 200 watts and is cooled by controlled radiation to the surroundings. The module weighs about 30 pounds and its dimensions are approximately 17 inches high by 7 inches square, including the mounting lugs. This module is the subject of a report to the Air Force Propulsion Laboratory, Wright-Patterson Air Force Base - "Orbital Fuel Cell," Technical Report AFAPL-TR-67-164, dated March 1968. This report is the culmination of about five years design and test effort under Air Force contract, including a successful orbital test launched by a Titan-3 booster, and results in a fuel cell system energy-to-weight ratio greater than 550 watt-hours/lb for an unmanned application.

The module is shown in Figure P-1. The horizontal rib-like structure is a set of louvers which regulate the module emissivity on three sides with respect to temperature. Module output characteristics are shown by the volt-ampere curves in Figure P-2. The 50 hour and 1014 hour curves represent net output current, whereas the 29 hour curve is gross output. Parasitic power consumed by heaters, purge controller, and current monitor circuits accounts for the difference. The module functional schematic is shown by the diagram in Figure P-3. Interfaces are with (1) the fuel cell electrical supervisory subsystem (ESS) which consists of purge controls and module switching, (2) telemetry, (3) the Reactant Supply Subsystem (RSS), (4) other fuel cell module plumbing, and (5) vehicle structure mounts. Module dimensions are shown by the drawings in Figure P-4. Reactant consumption versus gross power output is shown by the curves in Figure P-5 for 50 and 1014 hours of operation.

Modifications to the module for ELOR Lunar Module applications appear slight, but include control changes reflecting manned operation, which means reducing some of the automatic functions and the addition of cells to increase module voltage slightly. The module design has been tested for 1500 hours, so that no changes will be necessary to increase module life.

*Provided by Allis-Chalmers Div. of G.M. through Reference 4.17.

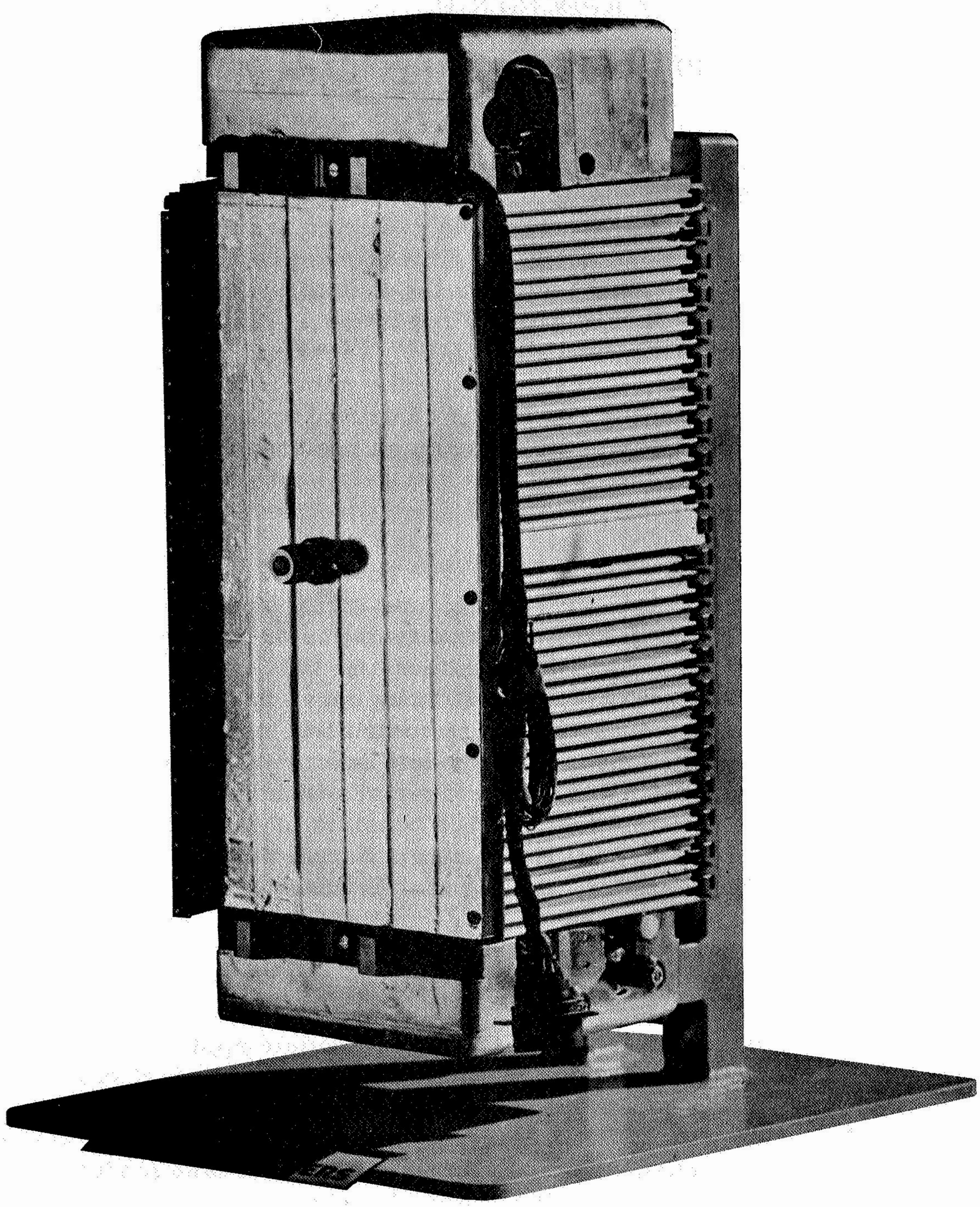

Figure P-1. Radiation Cooled Fuel Cell Module

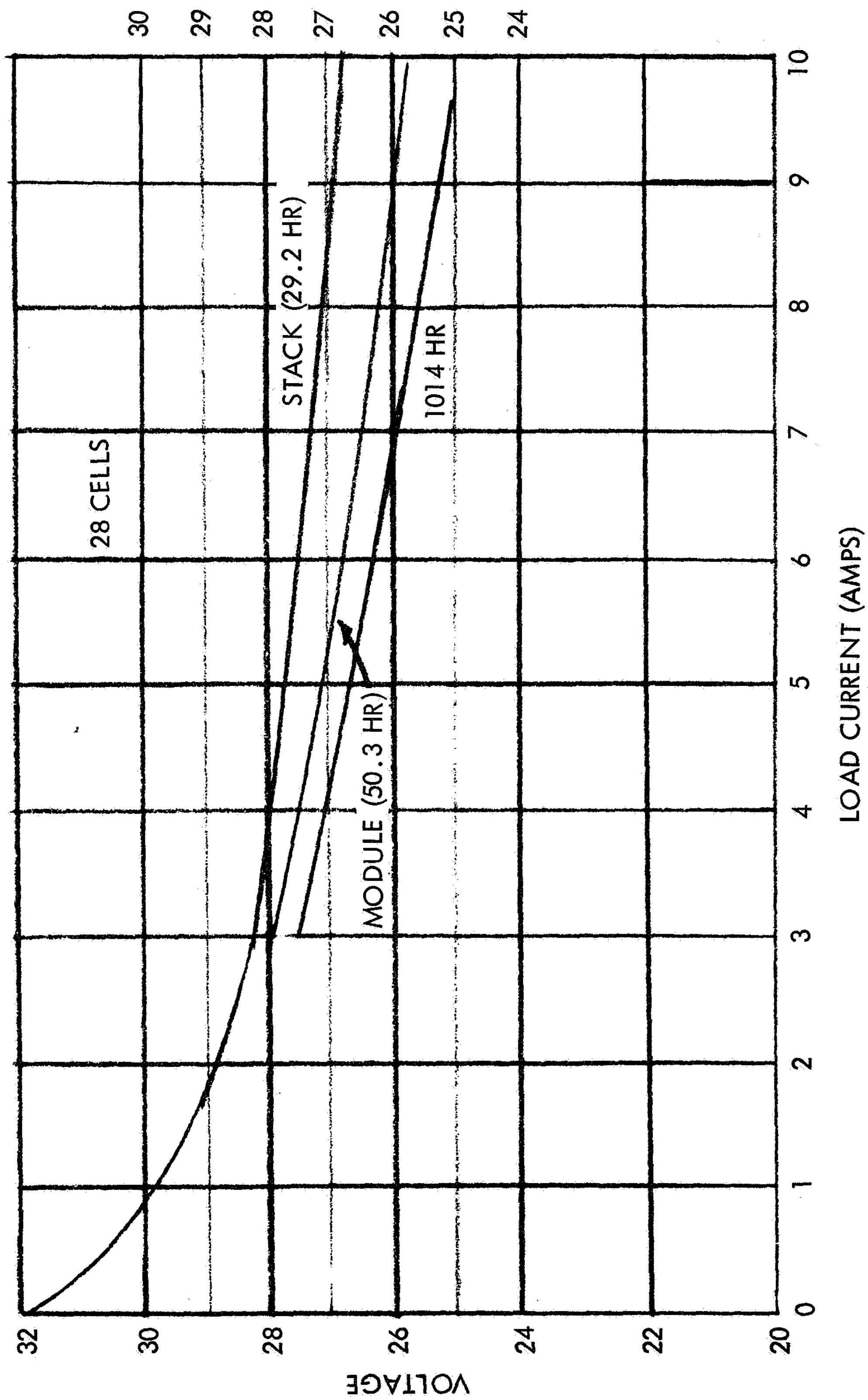

Figure P-2. Output Characteristics of Radiation-Cooled Module

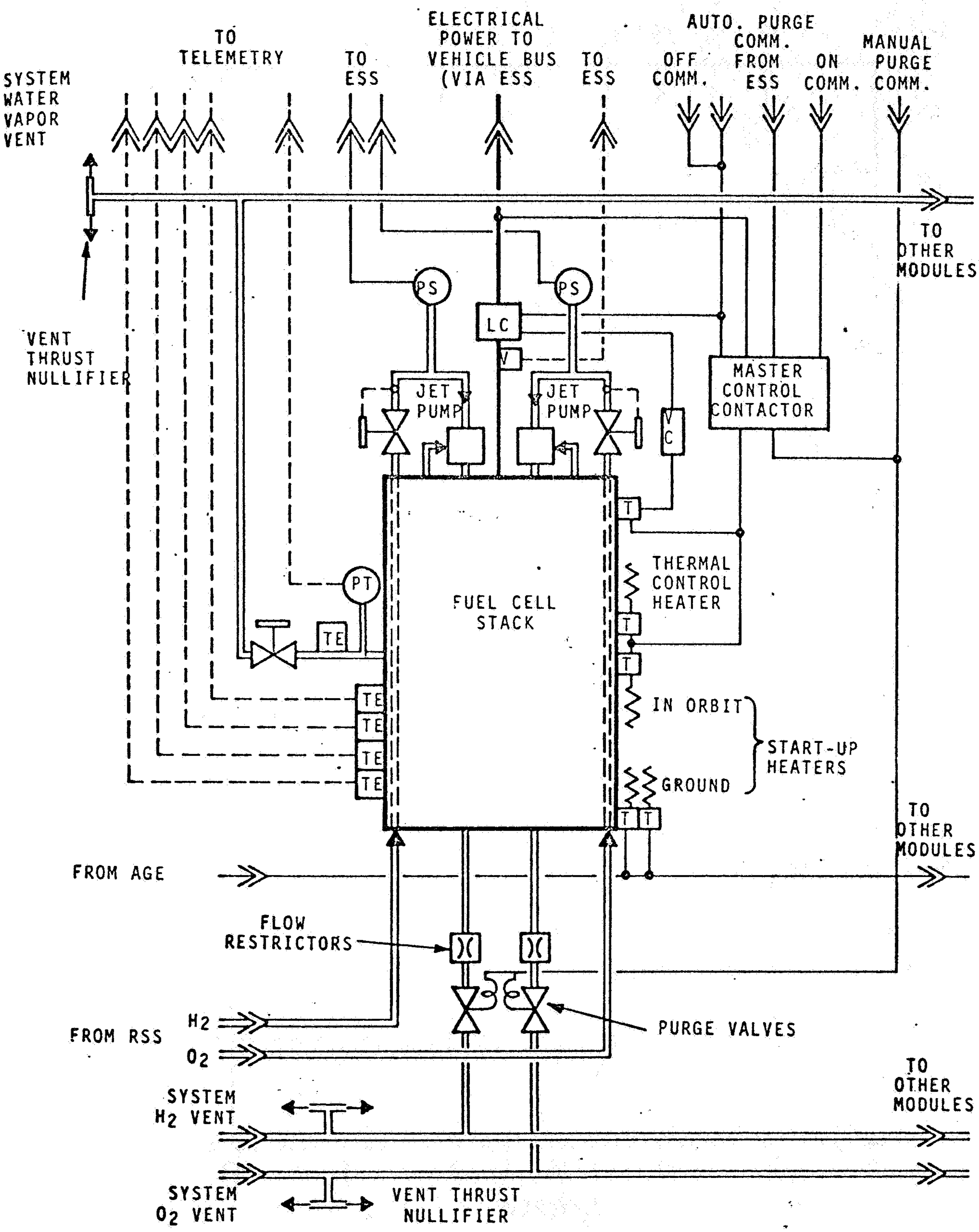

Figure P-3. Fuel Cell Module Subsystem Functional Schematic

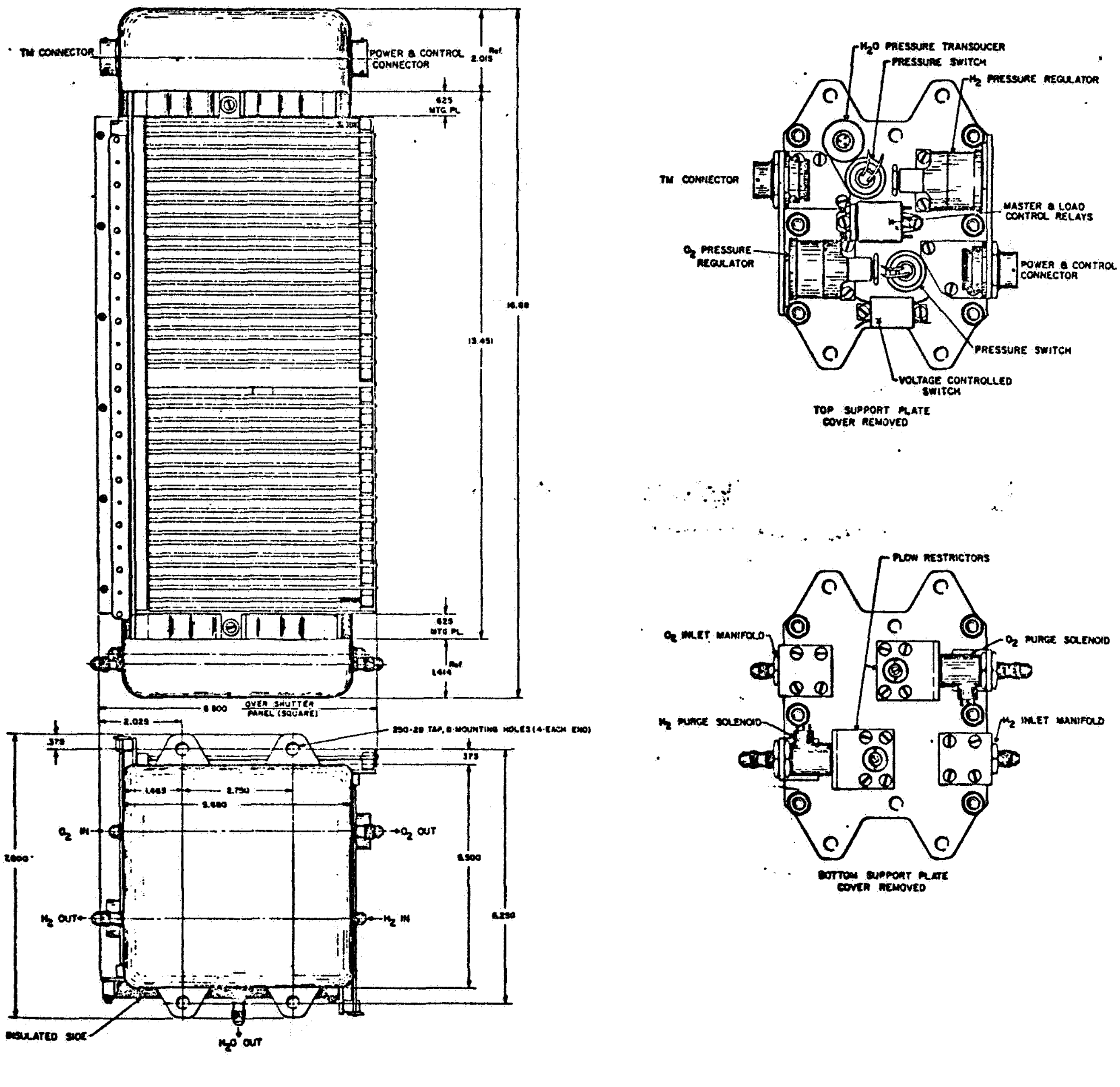

Figure P-4. Fuel Cell Stack Dimensions

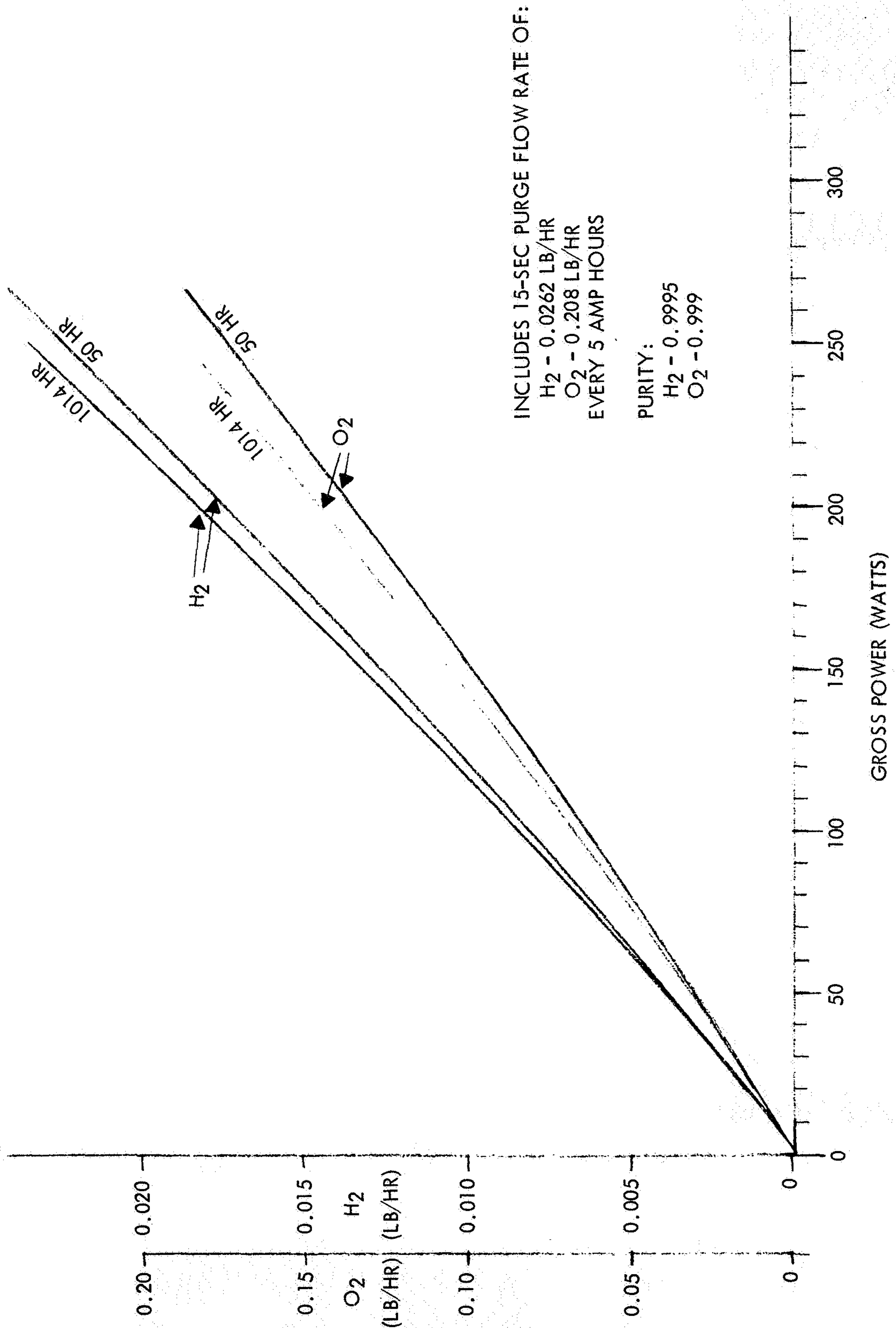

Figure P-5. Reactant Consumption for Orbital Fuel Cell Module No. 1, 28 Cells

APPENDIX Q

LM-ECS POWER AND WEIGHT SUMMARY, AS MODIFIED FOR ELOR (3-MAN)*

Table Q-1. Air Regeneration Subsystems

Item	No. Req'd	Peak Power (watts)	Avg. Power (watts)	Weight (lb)	Function
101	1			12.8	Cabin HX
102	2	225	73	1.4	Cabin Fan
106	1			2.08	Suit HX
107	1			8.12	Sublimator
109	2			2.21	Water Separator
110	1			4.34	LiOH Canister
111	1			0.78	Suit HX
112	1	16	0	2.13	Diverter Valve
113	1			0.93	Gas Check Valve
114	1			2.96	Diverter Valve
115	1			1.75	Diverter Valve
117	1			1.68	Relief Valve
118	2	585	156	4.81	Suit Fan
119	2			0.10	Check Valve
122	1			8.19	LiOH Cartridge
124	1			0.52	Pressure Sensor
125	1			4.43	LiOH Cartridge
127	1			2.82	LiOH Canister
134	1	4	0	0.88	Signal Control
138	3	33.5	0	3.51	Suit Circuit Valve
492	2			0.12	Check Valve
425-5	1			0.11	Flow Limiter
190 Pkg.	1			53.57	Suit Circuit Package
191 Pkg.	1			0.98	Cabin Fan Assembly
ARS Total				124.26	

*Data used herein was provided by the Hamilton Standard Division of UAC.

Table Q-2. Heat Transfer Subsystem

Item	No. Req'd	Peak Power (watts)	Avg. Power (watts)	Weight (lb)	Function
201	3	120	31.6	Part of 290	Pump and Motor
202	3			Part of 290	Relief Valve
203	1			1.09	Temp. Control Valve
204	1			2.52	Regenerative HX
208	1			0.28	Temp. Control Valve
209	1			15.41	Sublimator
210	1			3.48	Glycol Accumulator
212	1			Part of 290	Filter
213	3			Part of 290	Check Valve
214	2			0.09	Disconnect
218	1			290	P Sensor
222	1			0.24	Filter
223	2			0.18	Disconnect
224	1			4.11	Sublimator
290 Pkg.	1			8.22	Pump Package
HTS Total				35.89	

Table Q-3. Oxygen Supply System

Item	No. Req'd	Peak Power (watts)	Avg. Power (watts)	Weight (lb)	Function
304	4			0.30	Shutoff Valve
306	2			1.69	Regulator Valve
307	2			1.95	Cabin Dump Valve
309	1	22	0	1.46	Repressurization Valve
315	4			0.01	Filter
316	1			0.54	Oxygen Hose
323	1			1.37	Pressure Switch
390 Pkg.	1			12.98	Oxygen Supply Module
OSS Total				15.34	

Table Q-4. Water Management Subsystem (WMS)

Item	No. Req'd	Peak Power (watts)	Avg. Power (watts)	Weight (lb)	Function
401	6			0.01	Check Valve
402	2			0.16	Shutoff Valve
404	1			20.81	Water Tank Descent
406	1			0.12	Control Valve
409	2			5.31	Water Tank, Ascent
410	1			0.02	Isolation Valve
414	1			3.23	Selector Valve
415	3			0.55	Pressure Regulator
416	1			1.73	Water Hose
418	3			0.09	Disconnect
419	3			0.18	Disconnect
420	2			0.17	Shutoff Valve
425-1	1			0.09	Flow Limiter
425-3	1			0.11	Flow Limiter
490 Pkg.	1			6.12	Water Control Module
492	1			0.11	Check Valve Assembly
WMS Total				40.38	
Total System		1005.5	260.6	215.87	

APPENDIX R

LM-ECS STORAGE AND REACTIVATION PROCEDURE*

LM-ECS STORAGE REQUIREMENTS

The required storage conditions must be established prior to crew departure. System tests indicate that during the lunar day, maximum cabin temperature would be 96°F. This is well below the general specification temperature of 160°F maximum for component qualification, and below the boiling points of water and water/glycol solution at system pressures. The suit loop of the ARS should be kept above 3.0 psia in order to prevent boiling and thus loss of residual water in the water separators at temperatures below 140°F (an arbitrary selected value). Higher pressures increase the boiling point by only small increments. Achieving a satisfactory suit loop pressure could easily be accomplished by closing the loop and setting the two pressure-activated 306 regulator valves to their low pressure setting (3.8 ±0.2 psia). Leakage from this loop over a 90-day period should require less than 21.6 pounds of makeup oxygen assuming a leakage rate less than the present specification leakage rate for the suit loop of 0.01 pound per hour maximum.

A vacuum environment for the remainder of the cabin will not contribute significantly to offgassing. Lubricants are now available with sufficiently high molecular weights to be relatively unaffected by high or ultra-high vacuum environments. Other materials in the cabin should not be affected by high vacuum as the LM materials program includes testing for the selection only of materials suitable for high vacuum use. This program thus rejects materials with high offgassing capability or where material properties may be affected. It is recommended that the vehicle be closed in order to retain a slight vehicle pressure, thus retaining leakage losses from the pressurized suit loop in the cabin area. This will keep lunar dust out of the vehicle and exit hatch sealing surfaces.

It is noted that the item 420, 402 and 406 shutoff valves which supply water to the sublimators should be closed. This will lessen chances for accidental water loss from the sublimators. The porous plates of the sublimators are directly exposed to space vacuum and the lunar environment. Covers must be added to these sublimators to eliminate water loss by sublimation and to prohibit lunar dust from entering the porous plate of the

*Much of these data were provided by Hamilton Standard Division of UAC.

sublimator. The design of these covers will not be complex and only a small modification to the sublimators should be required, if any. The covers are estimated to weigh a total of 0.3 pound. They should be installed at the start of the lunar storage period and removed prior to the LM ascent mission phase.

In summary, it is recommended that during the quiescent mode, the LM ECS be secured except for maintaining a 3.8 psia oxygen atmosphere in the closed suit circuit. The item 306 oxygen pressure regulators should receive very few operating cycles to make up suit loop leakage losses, thus retaining nearly the same reliability for the ECS as the present LM mission. Cabin pressure may be essentially zero and temperature should be maintained above 32°F during lunar night periods. The LM water tanks are not located in the cabin area and require heating during lunar night periods. Trade-off studies may indicate a power saving by adding insulation to the water tanks and/or cabin walls versus additional heater power.

STORAGE MODE MONITORING REQUIREMENTS

During the 90-day storage mode, it is necessary to monitor the condition of the LM ECS and also the storage system. It is considered unnecessary to monitor equipment not in operation except where a gradual failure might be occurring that requires immediate corrective action or mission abort. Thus, it is desirable to monitor pressurized storage systems where leakage could jeopardize mission success and/or crew safety. Critical elements of the storage system must also be monitored including the performance of the storage system and state of the storage mode. Occasional operation of the ECS is not considered an appropriate means of status monitoring as startup and shutdown operations add severe cyclic mechanical, electrical and thermal demands on the system and increase chances for failure. Table R-1 indicates the storage-mode monitoring requirements.

Table R-1. Storage-Mode Monitoring Requirements

Data Required	Instrumentation Range
Suit Loop Pressure	2.0 - 6.0 psia
Suit Loop Temperature	0°F - 150°F
Cabin Temperature	0°F - 150°F
Cabin Pressure	0 - 6.0 psia
Descent Water Tank Temperature	30°F - 150°F
Ascent Water Tank Quantity (2)	0 - 50 lbs water
Cabin Wall Heater Power	0 - 75 watts
Descent Water Tank Power	0 - 10 watts
GOX Descent Tank Pressure	0 - 1200 psia
GOX Ascent Tank Pressure (2)	0 - 1200 psia

LM RCS REACTIVATION

Reactivation of the LM ECS following the 90 day storage period would be performed similar to the present LM system start-up. Disconnecting the power supply and communications cable to the lunar shelter will ultimately be necessary, but possibly should be delayed during the initial vehicle inspection. The suit loop circuit is pressurized but cannot be used until the ECS is fully in operation. The procedure for ECS activation will therefore include: activation of the coolant loop, sublimators and their water supply (the sublimator covers being removed during an initial external vehicle check), activation of the suit loop fan, and evaluation of system performance. Assuming one crew member has remained in the shelter until the LM vehicle is assured of flight readiness status, it is noted that the cabin must remain unpressurized until all crew members have transferred to the LM and are operating on the suit loop. This is required to allow Portable Life Support Systems (PLSS) sublimator operation which requires vacuum environment to reject heat. The astronauts will be wearing the PLSS during transfer and start-up.

For the activation of the LM ECS, an allowance of 60 minutes maximum should be made. This time considers the increased difficulty of operating from a pressurized space suit and PLSS, and also assumes the astronauts will be quite familiar with the ECS and activation procedures, but that reference to a check list will be required. A check of expendables and instrumentation will be included on the check list.

APPENDIX S

DETAILED LM VEHICLE ELOR TIME LINE*

In preparing an evaluation of the LM ECS requirements for the ELOR mission, it was necessary to construct a conservative time line estimate of the LM vehicle manned operations. The time line prepared is presented in Table S-1 for reference purposes. A number of the time estimates were derived from the present Apollo mission and should therefore be reasonable accurate. Those associated with the ELOR mission alone were conservative estimates only.

Table S-1. LM Manned Phase Operations, ELOR Mission

Mission Event	Duration (min)	Total Time (min)
LM Checkout	90.0	90.0
Separate	1.3	91.3
Orient LM	28.7	120.0
Insert into Syn Orbit	0.5	120.5
Coast to Pericynthion	60.0	180.5
Initial Powered Descent	6.0	186.5
Final Powered Descent	2.0	188.5
Hover to Touchdown	3.0	191.5
Post Landing & Prelaunch Checkout	75.0	266.5
Eat and Rest	60.0	326.5
PLSS Mission to Checkout External LM and to Checkout and Activate Shelter	300.0	626.5
Eat and Rest	360.0	986.5
PLSS Mission to Transfer two men to Shelter including Shelter Performance Monitor and Third man Don PLSS	180.0	1,146.5
Third Man Activate PLSS and Secure LM ECS	30.0	1,176.5
Actuate Storage System and Transfer to Shelter	30.0	1,206.5
Shelter Operations	90 Days	--

*Estimated by Hamilton Standard Corp. (Reference 4.32)

Table S-1. LM Manned-Phase Operations, ELOR Mission (Cont)

Mission Event	Duration (min)	Total Time (min)
START RETURN LEG		
Two Astronauts Transfer to LM and Inspect Vehicle Externally	60.0	60.0
Enter LM and Secure Storage System	5.0	65.0
Activate LM ECS and Checkout, Change to Suit Loop	60.0	125.0
Activate Other Systems and Checkout	30.0	155.0
Make Repairs and Replacement as Necessary	60.0	215.0
Third Astronaut Secure Shelter and Transfer to LM	30.0	245.0
Pressurize Cabin	5.0	250.0
Eat and Rest	60.0	310.0
Prelaunch Preparation (Descent)	70.0	380.0
Prelaunch Preparation (Ascent)	30.0	410.0
Powered Ascent	7.5	417.5
Orbit Contingency M-2A	30.0	447.5
Orbit Contingency M-2B	420.0	--
Orbit Contingency M-2C	90.0	--
Insert into Transfer Orbit	0.1	447.6
Coast in Transfer Orbit	90.0	537.6
Rendezvous with CSM	24.0	561.6
Dock	15.0	576.6
Crew Transfer	20.0	596.6

APPENDIX T

BANDWIDTH AND COMMUNICATION RANGE REQUIREMENTS, ELOR LM*

Table T-1. Summary of Bandwidth and Communication Range Requirements, ELOR LM

Mission Periods	Information Bandwidth and Subcarriers	Communication Range (nautical miles)
I. In Flight Periods		
A. VHF Links LM/CSM/LM - (Keyed Carrier Operation)		
1. Commands	1 to 2kb/s Max.	300 (Min) - 550 (Max)
2. Status Reception	1 to 2kb/s Max.	300 (Min) - 550 (Max)
3. Ranging	32 kHz Max (Fine Tone)	200 (Max)
4. PCM Data	1.6 kb/s	300 (Min) - 550 (Max)
B. S-Band with Earth (Identical with Basic Apollo)		
1. PCM TM Data	1.6 kb/s Min or 51.2 kb/s Max PSK on MHz S.C. 1.024 MHz S.C.	
2. Voice and Biomed	Voice (3KHz) + Biomed (on 14.5 kHz S.C.) on 1.24 MHz Subcarrier	

*Data used herein was provided by the RCA through Reference 4.34 of Vol. I.

Table T-1. Summary of Bandwidth and Communication Range Requirements, ELOR LM (Cont)

Mission Periods	Information Bandwidth and Subcarriers	Communication Range (nautical miles)
I. (Continued)		
B. (Continued)		
3. PRN Range Code	990 kb/sec	210,000
4. Up Data Link	1 kHz clock + 2 kHz PSK Tones on 70 kHz S. C.	
5. Up-Link Voice	3 kHz Max on 30 kHz S. C.	
6. Keyed Carrier	On - Off keying of 512 kHz S. C.	
7. Back up Voice	3 kHz Max	
II. <u>LUNAR STAY</u>		
A. <u>VHF Links LM/CSM/LM - (Keyed Carrier Operation)</u>		
1. Commands 2. Status Reception	1 to 2 kb/s Max 1 to 2 kb/s Max	300 (Min) to 550 (Max)
B. <u>VHF Links LM/EVA/LM - (Keyed Carrier)</u>		
1. Voice LM/EVA	3 kHz Max	1000 Ft. to shelter (But 119dB path loss capability)
2. Voice & EMU EVA to LM	Voice - 3 kHz + EMU on Sub-carriers of 3. 9, 5. 4, 7. 35 and 10. 5 kHz	Same as above

Table T-1. Summary of Bandwidth and Communication Range Requirements, ELOR LM (Cont)

Mission Periods	Information Bandwidth and Subcarriers	Communication Range (nautical miles)
II. (Continued)		
C. S-Band with Earth		
1. PCM TM Data during storage	1.5 kb/s on 1.024 KHz S.C.	
2. Voice reception for VHF relay LM/EVA for IIB above	3 kHz Max on 30 kHz S.C.	
3. Relay transmission of voice received EVA/LM on VHF per IIB above	3 mHz Max on 1.25 mHz S.C.	210,000
4. Up-data Link during depart preparations	2 kHz PSK Tones on 70 kHz S.C. + 1 kHz clock	
D. VHF Links LM/Shelter/LM - (Keyed Carrier)		
1. LM Status to Shelter	1 to 2 kb/s Max 1000 ft	1000 Ft.
2. Commands from Shelter (Optional)	1 to 2 kb/s Max	1000 Ft.

APPENDIX U

LM COMMUNICATIONS SYSTEM PERFORMANCE CAPABILITY AND DUTY CYCLE*

PERFORMANCE CAPABILITY

It is tentatively proposed that the LM CSS for the ELOR mission have the same performance capability as that provided for the basic Apollo DRM. That is, no changes are recommended at this time regarding transmitter power outputs, antenna gains, information bandwidths, receiver sensitivities, etc. This decision has been based on the assumption that the information bandwidths and communication range requirements are essentially the same as those for the DRM mission. (See Table T-1, Appendix T.) One possible exception here requiring further study relates to links between the LM and CSM when the LM is on the lunar surface. It is conceivable that the communication range and/or signal/noise requirements for the Command and Status Reception links may prove to exceed the existing VHF link capabilities. This could result in a need for either higher transmitter power output or a higher gain VHF antenna for placement on the lunar surface. (It should be noted that such an antenna was planned for the LM CSS early in the LM program, but this was eliminated in favor of use of the "In-Flight" omni type as the link requirements did not justify the increased gain.)

DURATION OF EQUIPMENT USAGE

Tables U-1, U-2, and U-3 summarize the anticipated hours of use of each of the components of the proposed LM Communications Subsystem for each of the information transfer links involved in the mission. The hours are further subdivided into In-Flight (IF) and Lunar Stay (LS) periods for each component, and periods of operate, standby and store. The duration periods shown in this table were based on the phase duration times given in Section 2 and a determination of the applicability of each of the components for the different links. Tables U-1, U-2, and U-3 apply, respectively, to Storage mode LM/Earth status data transmission alternatives A, B and C discussed previously.

*Much of the data used herein was provided by RCA.

It should be noted that the S-Band power amplifiers are utilized in the FULL OPERATE mode for only a few hours during the Lunar Stay phases of the mission. This is based on the assumption that the 1.6 kb/sec PCM data rate will be adequate for LM/Earth transmission of LM status data. This data rate can be successfully transmitted to earth with a bit error probability of less than 10^{-6} using the S-Band exciter level only (0.75 watt) and the S-band erectable antenna. The periods of operation shown for the Power Amplifiers during Lunar Stay relate to manned check-out (alternate C during storage phase) and manned usage for higher information bandwidths during the STORAGE PREPARATION and DEPART PREPARATION phases.

The hours of FULL OPERATE, STANDBY and STORE shown in Tables U-1, U-2, and U-3 were used in the preliminary reliability evaluation given in section 4.3.6 of this report.

Table U-1. LM Component Utilization per Communication Links - Alternative A

LM Communication Subsystem or EVC Component	Phase	Duration (Hours) in Conditions Indicated																							
		CSM Command Link			CSM Status Reception			LM/Earth Voice-Data Xmission			Earth/LM Reception (Voice Up-Data)			Earth/LM/EVA Voice and EVA/LM/Earth Voice-EMU			LM to Shelter Data			Earth/LM Up-Data Link (Via Shelter)			Totals		
		Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store
VHF ERA	IF																								
	LS																								
• VHF A Xmtr	IF	2.5	-	171.5										-		174							2.5	-	171.5
	LS	0.2	-	2193										2.0		2192							2.2		2192
• VHF A Rcvr	IF	Back-up to VHF B Rcvr for CSM Status and EVA Links																							
	LS																								
• VHF B Xmtr	IF																								
	LS																								
• VHF B Rcvr	IF	2.5	-	171.5	5.2	-	169							-		174							5.2		169
	LS	-	-	2193	6.0	-	2188							2.0		2192							8.0		2186
• Diplexer	IF	2.5	-	171.5	5.2	-	169							-		174							5.2		169
	LS	0.2	-	2193	6.0	-	2188							2.0		2192							8.0		2186
VHF Ant. Sel. Sw.	IF	2.5	-	171.5	5.2	-	169							-		174							5.2		169
	LS	0.2	-	2193	6.0	-	2188							2.0		2192							8.0		2186
VHF In-Flt. Ant. (1 or 2)	IF	2.5	-	171.5	5.2	-	169																5.2		169
	LS	0.2	-	2193	6.0	-	2188																6.0		2188
VHF EVA Antenna	IF													-		174							-		174
	LS													2.0		2192							2.0		2192
Signal Proc. ERA	IF																								
	LS																								
• Audio Center #1	IF							6.2		168	6.2		168	-		174							6.2		168
	LS							3.0		2191	3.0		2191	2.0		2192							5.0		2189
• Audio Center #2	IF							6.2		168	6.2		168	-		174							6.2		168
	LS							3.0		2191	3.0		2191	2.0		2192							5.0		2189
• PMP	IF							6.2		168	6.2		168	-		174							6.2		168
	LS							185		2009	3.0		2191	2.0		2192							187		2007
Comd. Mess. Gen. (1)	IF	2.5	-	171.5																			2.5		171.5
	LS	0.2	-	2193																			0.2		2193
Data Up-Link Assy	IF										6.2		168							-		174	6.2		168
	LS										3.0		2191							5.0		2189	8.0		2186
S-Band Tranceiver Assy	IF																								
	LS																								
• PM Mod (1 or 2)	IF							6.2		168				-		174							6.2		168
	LS							185		2009				2.0		2192							187		2007
• FM Mod.	IF	-																							
	LS																								
• Exciter (1 or 2)	IF							6.2		168				-		174							6.2		168
	LS							185		2009				2.0		2192							187		2007
• Receiver (1 or 2)	IF										6.2		168	-		174							6.2		168
	LS										3.0		2191	2.0		2192							5.0		2189

Table U-1. LM Component Utilization per Communication Links - Alternative A (Cont)

LM Communication Subsystem or EVC Component	Phase	Duration (Hours) in Conditions Indicated																							
		CSM Command Link			CSM Status Reception			LM/Earth Voice-Data Xmission			Earth/LM Reception (Voice Up-Data)			Earth/LM/EVA Voice and EVA/LM/Earth Voice-EMU			LM to Shelter Data			Earth/LM Up-Data Link (Via Shelter)			Totals		
		Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store
• Pwr Supply (1 or 2)	IF							6.2		168	6.2		168	-		174							6.2		168
	LS							185		2009	3.0		2191	2.0		2192							187		2007
S-Band Pwr. Amp. ERA	IF																								
	LS																								
• PA (1 or 2)	IF							6.2	0.2	168													6.2	0.2	168
	LS							3.0	0.1	2191													3.0	0.1	2191
• PA Pwr Sup. (1 or 2)	IF							6.2	0.2	168													6.2	0.2	168
	LS							3.0	0.1	2191													3.0	0.1	2191
• Diplexer	IF							6.2		168	6.2		168	-		174							6.2		168
	LS							185		2009	3.0		2191	2.0		2192							187		2007
S-Band Ant. Sel. Sw.	IF							6.2		168	6.2		168	-		174							6.2		168
	LS							185		2009	3.0		2191	2.0		2192							187		2007
S-Band Omni (1 or 2)	IF																								
	LS																								
S-Band Steerable Ant. Assy.	IF							6.2		168	6.2		168										6.2		168
	LS							3.0		2191	3.0		2191										3.0		2191
S-Band Erect. Antenna	IF							-		174				-		174							-		174
	LS							185		2009				2.0		2192							187		2007
Storage Data Coupler Assy.(1)	IF																-		174				-		174
	LS																92.5		2101				92.5		2101
Digital Ranging Gen. (DRG)	IF	2.5	-	171.5																			2.5		171.5
	LS	-	-	2194																			-		2194
CSM Status Monitor	IF				5.2		169																5.2		169
	LS				6.0		2188																6.0		2188
	IF																								
	LS																								
	IF																								
	LS																								
	IF																								
	LS																								
EVC (LM External)	IF																								
	LS																								
• EVC-1	IF													-		174							-		174
	LS													200		2192							200		1994
• EVC-1A	IF													-		174							-		174
	LS													200		2192							200		1994
• EVC-2	IF													-		174							-		174
	LS													200		2192							200		1994

Table U-2. LM CSS Component Utilization - Alternative B

(One 15 Min. Transmission LM/MSFN Each Day)

Note: This table is presented as a set of revisions to Table U-1. Use all values of Table U-1, except for the following revised hours for the components and link condition listed.

		LM/EARTH Voice - Data Xmission			TOTALS		
Component	Phase	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store
. PMP	LS	26		2168	28		2166
. PM MOD (1or2)	LS	26		2168	28		2166
. EXCITER (1or2)	LS	26		2168	28		2166
. PWR SUPPLY (1or2)	LS	26		2168	28		2166
. S-BAND DIPLEXER	LS	26		2168	28		2166
. S-BAND ANT. SEL SW.	LS	26		2168	28		2166
. S-BAND ERECT ANT.	LS	26		2168	28		2166

Table U-3. LM CSS Component Utilization - Alternative C

(4 Hours of operation 6 times during Lunar Stay Period)

NOTE: This table is presented as a set of revisions to Table U-1. Use all values of Table U-1 except the following revised hours for the components and link conditions listed.

Component	Phase	LM/EARTH Voice-Data Xmission			EARTH/LM Reception (Voice-Up Data)			TOTALS		
		Full Oper.	Stdby	Store	Full Oper.	Stdby	Store	Full Oper.	Stdby	Store
. Audio Center #1	LS	27		2167	27		2167	29		2165
. Audio Center #2	LS	27		2167	27		2167	29		2165
. PMP	LS	27		2167	27		2167	29		2165
Data Up-Link Assy.	LS				4		2190	9		2185
. PM MOD (1 or 2)	LS	27		2167				29		2165
. Exciter (1 or 2)	LS	27		2167				29		2165
. Receiver (1 or 2)	LS				27		2167	29		2165
. PWR Supply (1 or 2)	LS	27		2167	27		2167	29		2165
. PA (1 or 2)	LS	8.5	0.5	2185				8.5	0.5	2185
. PA PWR Sup. (1 or 2)	LS	8.5	0.5	2185				8.5	0.5	2185
. S-Band Diplexer	LS	27		2167	27		2167	29		2165
S-Band Ant. Sel SW	LS	27		2167	27		2167	29		2165
S-Band Steerable Ant.	LS	9		2185	9		2185	9		2185
S-Band Erect. Ant.	LS	21		2173	21		2173	23		2171

APPENDIX V

SPARES REQUIREMENTS ANALYSIS FOR THE LM COMMUNICATIONS SYSTEM*

The approach applied in this study is to provide sufficient failure detection and spares so that the probability of successful lunar stay and return is not reduced by the extended ELOR lunar stay as compared with the LOR lunar stay.

Thus, the sum of the equipment failure probabilities during lunar stay is calculated (see Table V-1). This sum is used as the objective for the ELOR lunar stay, and is to be accomplished by providing spares for the higher failure probability units.

ALTERNATIVE A

From Table V-1 we note that the probability of loss of any LM CSS capability during LOR lunar stay is:

$$P_{LOR} = 0.0045$$

The corresponding value for the ELOR mission with no spares is:

$$P_{ELOR} = 0.0171$$

Now P_{ELOR} must be reduced to, or less than P_{LOR}, by the use of spares. From Table A-22 we see that each of the 3 EVC units has a failure probability during Lunar Stay of 0.0038. The EVC units provide an essential lunar operation capability and, like the Signal Processor Assembly, they also have the highest failure probabilities. If a spare EVC-2 is provided, Prob of loss of EVC-2 Capability = $(0.0038)^2 = 0$. If a spare EVC-1 is also included, (note that this can also be used for EVA-1A) we have Prob. of loss of either EVC-1 or EVC-1A capability is equal to:

$$(0.0038)^3 + 3\ (0.0038)^2\ (0.9962) = 0.$$

Thus, providing both spare EVC-1 and EVC-2 units results in a probability reduction of:

$$3\ (0.0038) = 0.0114.$$

*Provided by the RCA through Reference 4.34 of Vol. I.

But a further reduction of 0.0012 is required to reach with 0.0045 goal. The only unit having a failure probability this great or greater is the Signal Processor. This unit is also essential for both lunar stay and in-flight operation.

Prob. loss of SPA = 0.0040

If a spare is provided

Prob. loss of SPA = $(0.0040)^2$ = 0.

Thus, with spare EVC-1, EVC-2 and SPA units, we have:

Prob. of loss of any LM CSS capability during ELOR Lunar Stay is:

$$P = 0.0170, - 0.0114 - 0.004 = 0.0017.$$

This is less than the probability of loss of any LM SCC capability during LOR lunar stay. Thus, the recommended spares for the alternative A mission are EVC-1, EVC-2 and SPA; one each.

ALTERNATIVE B

From Table V-1:

$$P_{LOR} = 0.0045$$

$$P_{ELOR} = 0.0137$$

By using the same approach delineated under alternative A above, the resultant spare recommendations are one EVC-1 and one EVC-2. The resultant probability of failure is 0.0023, or less than the LOR mission value.

ALTERNATIVE C

From Table V-1:

$$P_{LOR} = 0.0045$$

$$P_{ELOR} = 0.0147$$

The resultant recommendations are the same as for alternative B; namely one EVC-1 and one EVC-2 spare. The resultant probability of

Table V-1. Probability of Loss of LM Capability During Lunar Stay

LM Communication Subsystem or EVC Component (per ERA)	LOR MISSION							ELOR "A" MISSION						
	Energized State			Non-Energized State			ERA Totals $\lambda_E T_E$ Plus $\lambda_N T_N$	Energized State			Non-Energized State			ERA Totals $\lambda_E T_E$ Plus $\lambda_N T_N$
	λ_E (F/10^6)	T_E (Hrs)	$\lambda_E T_E$ (x10^{-6})	λ_N (F/10^6)	T_N (Hrs)	$\lambda_N T_N$ (x10^{-6})		λ_E (F/10^6)	T_E (Hrs)	$\lambda_E T_E$ (x10^6)	λ_N (F/10^6)	T_N (Hrs)	$\lambda_N T_N$ (x10^6)	
VHF ERA														
• VHF A Transmitter	13.0	35	455.0	0.013	0	0		13.0	2.2	28.6	0.013	2191.8	28.49	
• VHF A Receiver	11.0	35	385.0	0.011	0	0		11.0	0	0	0.011	2194	24.13	
• VHF B Transmitter	12.0	0	<0.001	0.012	35	0.42	810.805	12.0	0	0	0.012	2194	26.33	219.600
• VHF B Receiver	11.0	3	<0.001	0.011	35	0.385		11.0	8	88.0	0.011	2186	24.05	
• Diplexer	-	-	-	-	-	-	(Failure Rate Included Above)			-	-	-	-	-
VHF Antenna Select Switch	0.2	1	0.2	0.020	34	0.68	0.868	0.2	8	1.6	0.020	2186	43.72	45.320
VHF In-Flight Ant (1 or 2)	0.1	1	<0.001	0.010	34	0.00	0.000	0.1	6	0	0.010	2188	0.05	0.048
VHF EVA Antenna	0.08	35	2.8	0.010	0	0.00	2.800	0.08	2	0.2	0.010	2192	21.92	22.100
Signal Processor ERA														
• Audio Center #1	19.0	35	665.0	0.019	0	0.00		19.0	5	95.0	0.019	2189	41.60	
• Audio Center #2	14.0	35	490.0	0.014	0	0.00	1820.000	14.0	5	70.0	0.014	2189	36.65	1024.370
• PMP	19.0	35	665.0	0.019	0	0.00		19.0	187	3743.0	0.019	2007	38.13	
Command Message Gen.			(Not Included in LOR Mission)				-	51.0	0.2	10.2	0.051	2193.8	111.85	122.084
CSM Status Monitor			(Not Included in LOR Mission)				-	15.0	6	90.0	0.015	2188	32.82	122.820
S-Band Transceiver ERA														
• PM Mod (1 or 2)	4.0	35	0.02	0.004	0	0.00		4.0	187	0.6	0.004	2007	<<0.01	
• FM Mod	11.0	35	385.0	0.011	0	0.00		11.0	5	55.0	0.011	2189	24.08	
• Exciter (1 or 2)	7.0	35	0.06	0.007	0	0.00	386.260	7.0	187	1.7	0.007	2007	<<0.01	83.620
• Receiver (1 or 2)	30.0	35	1.10	0.030	0	0.00		30.0	5	<<0.1	0.030	2189	<<0.01	
• Power Supply (1 or 2)	8.0	35	0.08	0.008	0	0.00		8.0	1.87	2.2	0.008	2007	<<0.01	
S-Band Power Amp. ERA														
• Power Amp (1 or 2)	30.0	3	0.01	0.030	32	0.96		30.0	3	<<0.1	0.030	2191	<<0.01	
• Power Supply (1 or 2)	10.0	3	0.01	0.010	32	0.32	29.290	10.0	3	<<0.1	0.010	2191	<<0.01	169.684
• Diplexer	0.8	35	28.00	0.010	0	0.00		0.8	187	149.6	0.010	2007	20.07	
S-Band Antenna Select Switch	0.2	35	7.00	0.020	0	0.00	7.000	0.2	187	37.4	0.020	2007	40.14	77.540
S-Band Omni Ant. (1 or 2)	0.1	1	0.10	0.010	34	0.34	0.440	0.1	1	1	0.010	2193	21.93	22.030
S-Band Steerable Ant.	22.0	3	66.00	0.020	32	0.64	66.640	22.0	3	66.0	0.020	2191	43.82	109.820
S-Band Erectable Ant.	0.5	32	16.00	0.010	3	0.30	16.030	0.5	187	93.5	0.010	2007	20.07	113.570
Digital Ranging Gen.	18.0	0	0.00	0.018	35	0.63	0.630	18.0	0	0	0.018	2194	39.49	39.492
Store Mon/Alarm & Program Unit			(Not Included in LOR Mission)				-	4.0	93	372.0	0.040	2101	84.04	456.040
EVC (LM External)														
• EVC-1	19.0	35	665.00	0.020	0	0.00	665.000	19.0	200	3800.0	0.020	1994	39.88	3839.880
• EVC-1A			(Not Included in LOR Mission)				-	19.0	200	3800.0	0.020	1994	39.88	3839.880
• EVC-2	20.0	35	700.000	0.020	0	0.00	700.000	20.0	200	4000.0	0.020	1994	39.88	3839.880
Totals							4505.763							17147.778

Table V-1. Probability of Loss of LM Capability During Lunar Stay (Cont)

LM Communication Subsystem or EVC Component (per ERA)	ELOR "B" MISSION							ELOR "C" MISSION						
	Energized State			Non-Energized State			ERA Totals $\lambda_E T_E$ Plus $\lambda_N T_N$	Energized State			Non-Energized State			ERA Totals $\lambda_E T_E$ Plus $\lambda_N T_N$
	λ_E (F/10^6)	T_E (Hrs)	$\lambda_E T_E$ (x10^{-6})	λ_N (F/10^6)	T_N (Hrs)	$\lambda_N T_N$ (x10^{-6})		λ_E (F/10^6)	T_E (Hrs)	$\lambda_E T_E$ (x10^6)	λ_N (F/10^6)	T_N (Hrs)	$\lambda_N T_N$ (x10^6)	
VHF ERA														
• VHF A Transmitter	13.0	2.2	28.6	0.013	2191.8	28.49		13.0	2.2	28.6	0.013	2191.8	28.49	
• VHF A Receiver	11.0	0	0	0.011	2194	24.13		11.0	0	0	0.011	2194	24.13	
• VHF B Transmitter	12.0	0	0	0.012	2194	26.33	219.600	12.0	0	0	0.012	2194	26.33	219.600
• VHF B Receiver	11.0	8	88.0	0.011	2186	24.05		11.0	8	88.0	0.011	2186	24.05	
• Diplexer	-	-	-	-	-	-	(Failure Rate Included Above)			-	-	-	-	
VHF Antenna Select. Switch	0.2	8	1.6	0.020	2186	43.72	45.320	0.2	8	1.6	0.020	2186	43.72	45.320
VHF In-Flight Ant. (1 or 2)	0.1	6	<<0.1	0.010	2188	0.05	0.001	0.1	6	0	0.010	2188	0.05	0.048
VHF EVA Antenna	0.08	2	0.2	0.010	2192	21.92	22.100	0.08	2	0.2	0.010	2192	21.92	22.100
Signal Processor ERA														
• Audio Center #1	19.0	5	95.0	0.019	2189	41.60		19.0	29	551.0	0.019	2165	41.14	
• Audio Center #2	14.0	5	70.0	0.014	2189	36.65	816.400	14.0	29	406.0	0.014	2165	30.31	1620.580
• PMP	19.0	28	532.0	0.019	2166	28.16		19.0	29	551.0	0.019	2165	41.14	
Command Message Gen.	51.0	0.2	10.2	0.051	2193.8	111.85	122.080	51.0	0.2	10.2	0.051	2193.8	111.85	122.084
CSM Status Monitor	15.0	6	90.0	0.015	2188	32.82	122.820	15.0	6	90.0	0.015	2188	32.82	122.820
S-Band Transceiver ERA														
• PM Mod. (1 or 2)	4.0	28	<<0.1	0.004	2166	<0.01		4.0	29	<<0.1	0.004	2165	<<0.01	
• FM Mod.	11.0	5	55.0	0.011	2189	24.08		11.0	5	55.0	0.011	2189	24.08	
• Exciter (1 or 2)	7.0	28	<<0.1	0.007	2166	<<0.01	79.080	7.0	29	<<0.1	0.007	2165	<<0.01	79.880
• Receiver (1 or 2)	30.0	5	<<0.1	0.030	2189	<<0.01		30.0	29	0.8	0.030	2165	<<0.01	
• Power Supply (1 or 2)	8.0	28	0.1	0.008	2166	<<0.01		8.0	29	<<0.1	0.008	2165	<<0.01	
S-Band Power Amp. ERA														
• Power Amp (1 or 2)	30.0	3	<<0.1	0.030	2191	<<0.01		30.0	9	<<0.1	0.030	2185	<<0.01	
• Power Supply (1 or 2)	10.0	3	<<0.1	0.010	2191	<<0.01	44.060	10.0	9	<<0.1	0.010	2185	<<0.01	44.936
• Diplexer	0.8	28	22.4	0.010	2166	21.66		0.08	29	23.2	0.010	2165	21.65	
S-Band Antenna Select. Switch	0.2	28	5.6	0.020	2166	43.32	48.920	0.02	29	5.8	0.020	2165	43.30	49.100
S-Band Omni Ant. (1 or 2)	0.1	1	0.1	0.010	2193	21.93	22.030	0.1	1	0.1	0.010	2193	21.93	22.030
S-Band Steerable Ant.	22.0	3	66.0	0.020	2191	43.82	109.820	22.0	9	198.0	0.020	2185	43.70	241.700
S-Band Erectable Ant.	0.5	28	<<0.1	0.010	2166	<0.01	0.001	5.0	23	115.0	0.010	2171	21.71	136.700
Digital Ranging Gen.	18.0	0	0	0.018	2194	39.49	39.492	18.0	0	0	0.018	2194	39.49	39.492
Store Mon/Alarm & Program Unit	4.0	93	372	0.040	2101	84.04	456.040	4.0	93	372	0.040	2101	84.04	456.040
EVC (LM External)														
• EVC-1	19.0	200	3800.0	0.020	1994	39.88	3839.880	19.0	200	3800.0	0.020	1994	39.88	3839.880
• EVC-1A	19.0	200	3800.0	0.020	1994	39.88	3839.880	19.0	200	3800.0	0.020	1994	39.88	3839.880
• EVC-2	20.0	200	4000.0	0.020	1994	39.88	3839.880	20.0	200	4000.0	0.020	1994	39.88	3839.880
Totals							13673.004							14742.070

failure is 0.0033; again less than the LOR mission. It should be noted, for alternatives B and C, that the SPA failure probability is much lower than for alternative A due to the shorter time duration in which it is energized.

APPENDIX W

DETAILS OF CM EARTH LANDING SYSTEM (ELS) ANALYSIS FOR ELOR MISSIONS*

POTENTIAL PROBLEM AREAS

Extended exposure to the lunar orbital environment requires analysis of the anticipated effects on the earth landing system. Specific problem areas which require examination include:

1. The possibility that exposure to mission temperature and pressure environments may degrade the system textiles to an extent that the system performance is degraded.

2. The possibility of cold welding of metallic components.

3. The possibility of the degradation of pyrotechnics from exposure to the combined environments.

4. The possibility of functional degradation of plastics and elastomers in the system from exposure to the thermal/vacuum environment.

5. The possibility of system degradation from penetrating radiation and meteoroids.

LIFE ANALYSIS

Effects of Storage Temperatures

During lunar coast, lunar orbit and earth coast the temperature of the ELS components is expected to be between a maximum of 120°F and a minimum of -65°F. The exception is the sequence controller, which is located within the crew compartment where temperatures will be maintained within the range of 32 to 120°F (Ref. 2). During these mission phases the landing system is in a storage, rather than an operational mode.

As shown in Table W-1, the temperature extremes are within published "continuous service" temperature ranges for materials used in the landing

*Much of the data used herein was provided by the Ventura Division of Northrop Corp.

Table W-1. Continuous Service Temperature

	Minimum (°F)	Maximum (°F)	References
Nylon	-320	175	12, 21
Dacron	-423	250	12, 21
Cotton	-70	275	5, 15
Nomex	-320	350	12, 30
Teflon (polytetrafluoroethylene)	-423	500	12
Polyester resin-glass fiber	-423	250	12
Irradiated polyethylene	-67	275	14
Polyurethane foam	-423	300	12
Viton O-ring	-65	340	12
Lubeco 905 solid lubricant	-452	500	13
Aluminum	-423	700	12
Stainless steel	-423	1500	12

system. However, possible system degradation because of the in-flight storage temperature will be examined further.

The system components which are the most vulnerable to the minimum temperature of -65°F are probably the Viton O-rings and seals used in the mortar assemblies. At very low temperatures this material becomes hard and brittle and has very little impact strength. Cracking would allow the escape of gases when the pressure cartridges are fired, thus degrading the effective output of the pressure cartridges. However, Viton has been successfully used for static seals at temperatures well below -65°F (Ref. 12), and it can be expected that both the Viton O-rings and seals are capable of withstanding non-operational storage at -65°F for the duration of the mission. Temperatures lower than the specification minimum temperature are undesirable. However, an absolute minimum acceptable temperature cannot be established on the basis of available information.

At the upper end of the specification temperature range, the main consideration is the effect on the useful service life of the system components. The duration of the useful life of all organic materials is highly dependent on temperature, with the useful lifetime decreasing rapidly as temperature increases. As an approximation, useful life can, in general, be expected to be cut in half for each 18°F increase in storage or service temperature.* Or, more accurately, the rate of degradation in properties can be expressed by the Arrhenius equation

$$k = A \exp(-E/RT),$$

where k is the rate of degradation at the absolute temperature T (A and E are constants for a given material and R is a constant).

In terms of a capability to withstand continuous storage at the maximum temperature of 120°F, the pyrotechnics and the nylon textiles will be specifically considered. As shown in Table W-2, the pyrotechnics can be expected to withstand continuous storage at 120°F for the duration of the ELOR Mission.

An analysis of data published by duPont (Ref. 21) indicates that textiles made from heat resistant types of bright high tenacity nylon 66 (such as Types 330 or 700 or the equivalent) can withstand continuous exposure to a temperature of 120°F in air for 99 days with no significant degradation. However, if the temperature were raised an additional 20°F or if non-heat resistant nylon types (such as Type 300) were used, measurable degradation would be expected (in air) after this period of time.

The rate of the thermal degradation of nylon, as well as many other organic polymers, is much higher in the presence of oxygen than in the absence of oxygen. Thus, the thermal stability of nylon would be greater in the ELS compartment during the lunar orbital phase than under normal atmospheric conditions. Continuous exposure to the maximum specification temperature for the duration of the mission is expected to have no deleterious effect on the nylon textiles. However, temperatures in excess of the maximum specification temperature for significant periods of time would be undesirable.

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*Degradation in physical properties due to heat is basically related to chemical degradation of the organic material. Thus, it can be expected to follow the rule of thumb that the rate of a chemical reaction doubles with each 18°F increase in temperature.

Table W-2. Shelf Life of Pyrotechnics

Component	Continuous Temperature of					Ambient or Cyclic Conditions			Ref.
	150°F	160°F	165°F	167°F	200°F	-60 to 165°F	25 to 105°F	-20 to 110°F	
Pressure Cartridge							4 yrs(3)		6
Hercules "Hi Temp"	Indefinite							20 yrs	16
Hercules "5250.95"	500 days							20 yrs	16
Reefing Line Cutter							4 yrs(3)		6
M-42 Primer				5 yrs(1)	28 days(2)				-
A1A Ignition Mix		1 yr				10-15 yrs			19
Mil-C-13739A Delay Mix			1 yr						20
Hercules "unique"	250 days							20 yrs	16

(1) Reference 17

(2) Reference 18

(3) Specification requirements (Reference 6)

Effects of Storage Pressures

The present Apollo specification pressure for the ELS compartment is 10^{-6} torr. At low pressure various effects on the materials used in the landing system are possible, such as (a) loss of moisture and gases normally present on and within materials, (b) loss of materials by vaporization or decomposition, and (c) cold welding of contacting metal surfaces.

In the absence of a surrounding cloud of gas molecules, the evaporation or sublimation of materials can be described by the Langmuir equation:

$$W = PA \frac{M}{2\ RT}$$

where W, P, A, and M are the weight evaporated per unit time, the vapor pressure, surface area and molecular weight of the material. R is the gas constant and T the absolute temperature. Evaportation rate increases rapidly with temperature because the vapor pressure term increases rapidly as temperature increases.

In an enclosed space, such as the ELS compartments, a finite atmosphere of evaporated material will accumulate. If the partial pressure of a material as a gas equals the vapor pressure of the material at the current temperature, then apparent (or net) evaporation will cease. When some evaporated material is present as a vapor then the evaporation rate can be expressed by substituting (P_2 - P_1) for P in the previous equation. P_2 is the vapor pressure of the material, P_1 the partial pressure of the material in the gas phase. The presence of other types of molecules in the vapor state would also decrease the net rate of evaporation of a given material by "bouncing" some of the evaporating molecules back to the solid surface.

At the low pressures in the ELS compartment the mean free path of molecules in the gas phase can be expected to exceed the dimensions of the compartment. A molecule which escapes a surface will usually continue going until it strikes another surface and either sticks or bounces off. There will be a tendency for any material which evaporates to eventually cover all surfaces in the compartment and to evaporate from these surfaces at a finite rate. There will be a tendency for material which evaporates to accumulate on any cooler surfaces in the compartment.

The ELS compartment is vented. Thus, there will be a rapid loss of air from the compartment during the launch phase of the mission, followed by a slow loss of material. When the mean free path of molecules within the compartment equals or exceeds the diameter of the venting orifices, then a general expression for the rate of loss of material from the compartment is the same as the equation given previously, with (P_2 - P_1) substituted for P. In this case P_2 is the pressure inside the compartment, P_1 is the pressure outside the compartment.

Metallic Components

The evaporation rates for metals used in the ELS is very low. After 99 days at temperatures up to 120°F the loss of material will be insignificant for the metals used in the landing system (Ref. 5).

Under proper laboratory conditions, cold welding of contacting clean metal surfaces has been reported. However, tests in which polished and non-polished specimens of stainless steel, aluminum alloy and titanium alloy (6 percent aluminum, 4 percent vanadium) were held in contact in a vacuum of 3×10^{-6} torr at room temperature for 1000 hours failed to demonstrate cold welding (Ref. 22). Colding welding of the metals used in the landing system requires the contact of perfectly clean bare metal surfaces. It is favored by (and generally requires) a combination of temperatures much higher than 120°F, high contact pressure, and a rubbing motion. It is less probable with dissimilar metals than with similar metals in contact.

In the ELS, critical metal to metal contacts are:

1. The interface between mortar covers and mortar tubes
2. The strands in the stainless steel riser cables
3. The swivel fittings at the riser attachments to the flowerpot on the command module.

Mortar tubes and covers are made of anodized aluminum. Although anodized coatings may be altered somewhat by dehydration in vacuum, this will have no adverse effect on the system. The aluminum oxide coating itself is very stable in vacuum. Its decomposition pressure is extremely low and its vapor pressure is lower than that of aluminum. Motion between the mortar cover and mortar tube is prevented by a silicone moisture sealant. It is expected that there will be no significant loss of the oxide layer of aluminum surfaces and that cold welding will not occur during the Extended Lunar Orbital Mission. Riser cable strands are protected by an inorganic vacuum-stable dry film lubricant. This can be expected to provide adequate protection from cold welding of the cable strands.

The "flowerpot" on the command module is a titanium alloy. The riser attachment is stainless steel. The dissimilar contacting metals can be expected to discourage cold welding. In addition, in experiments in which cold welding of titanium has a tenacious oxide film that is difficult to remove. (Adhesion of titanium to titanium was produced by careful surface cleaning and polishing and application of a high load together with a rotary motion in a vacuum of 5×10^{-10} torr — and then only if the temperature was 302 degrees or higher.)

Plastic and Elastomers

The non-metallic components in the ELS are primarily organic polymers. In general, the basic polymer has a very high molecular weight and a very low vapor pressure. In a vacuum the vaporization rate for the basic polymer can generally be considered to be negligible, as long as the temperature is not so high that thermal degradation is taking place. However, polymers "outgas" and loose weight in a vacuum, with the rate and amount depending on the type of material, how it was processed and the temperature. The rate of weight loss usually becomes progressively less with time. Weight loss can not necessarily be equated with loss or degradation of functional capability.

Table W-3 presents a summary of the vacuum effects on plastics and elastomers used in the present Block II Increased Capability Earth Landing System. As shown in the table, RTV-102 silicone resin loses a considerable amount of volatile material. The rate of material loss can be expected to decrease with time. Loss of lower molecular weight fractions and residual starting materials is generally to be expected with silicone resins and, as used in the landing system would not be expected to have an adverse functional effect on the system. The available information indicates the plastics and elastomers used in the ELS will be capable of serving their intended function at the conclusion of the ELOR Mission.

Textiles

The textiles used in the ELS are organic polymers. General statements made in the section on plastics and elastomers (4.2.2.2) apply equally well to these textiles.

Nylon is used as the major load bearing structural textile in the earth landing system. A selection of vacuum test data is summarized in Table W-4. The data tends to indicate there may be a small decrease in nylon strength following vacuum exposure, with no apparent relationship between length of vacuum exposure and change in strength. Except for a reported 10 percent decrease in nylon fabric strength 25 minutes after vacuum exposure, the strength changes shown in Table W-4 are generally within the normal scatter expected for textile tests. Nylon textiles have been used successfully in landing systems for numerous space missions, and available evidence appears to support a conclusion that the basic nylon material is quite stable in a

Table W-3. Vacuum Stability of Plastics and Elastomers Used in the Apollo Landing System

Material	Application	Vacuum Reliability	Ref.
Viton (hexafluoropropylene-vinylidine copolomer)	O-rings and molded seals in mortars	1) Retains excellent properties after storage for nine months at 10^{-6} torr and 257°F 2) A preferred material for vacuum sealing	24
Fiberglass reinforced polyester	Sabot in mortars	Satisfactory	12
RNF-100 Thermofit Tube (irradiated polyethylene)	Protects steel riser cables from abrasion during deployment	0.62% weight loss after 336 hrs at 10^{-6} torr and 275°F	24
Polyurethane foam (rigid closed cell foam)	Immobilizes steel riser cables	Good vacuum stability	12
RTV-102 Silicone Resin	Moisture seal between mortar cover and mortar tube during ground storage (not intended as a vacuum seal)	3% weight loss after 24 hrs at 10^{-6} torr and 257°F	24
Polytetrafluoroethylene (Teflon)	Buffer in parachute pack	Good	12

vacuum environment. However, because of the limited amount of experiment and theoretical evidence available a 10 percent reduction in the strength of all textiles has been assumed in the reliability analysis.

Other textiles in the ELS do not share the critical load bearing requirement with nylon. Available information (Table W-4) indicates they will be functionally adequate at the conclusion of the ELOR Mission.

Table W-4. Effects of Vacuum Exposure of Textiles

Type of Textile	Exposure Conditions	Test Conditions	Time After Removed from Vacuum (min)	Change in Tensile Strength (%)	Ref.
Nylon Fabric 1.1 oz/yd^2 (Mil-C-7020 Type 1)	About 30 days, 1×10^{-6} torr 75°F	Atmospheric pressure 75°F 60% R.H.	5 25 40	-2 -10 -4	23
Nylon Monofilament 15 denier Type 330 nylon	145 days 4×10^{-8} torr room temp	Atmospheric pressure 50-67% R.H.	2 4 6 8 10 12 15	-1.7 -2.6 +1.7 +0.3 -1.3 -0.9 -2.1	26
Nylon Fabric 2.0 oz/yd^2 (Mil-C-7350-I)	64 hrs 1×10^{-6} torr 70°F	2.8×10^{-5} torr 70°F	-	-2.8	27
Dacron Fabric 1.7 oz/yd^2 2 x 1 twill	64 hrs 1×10^{-6} torr 70°F	2.8×10^{-5} torr 70°F	-	+1.1	27
Nomex Fabric 2.0 oz/yd^2 Mil-C-7350-I	64 hrs 1×10^{-6} torr	2.9×10^{-5} torr 70°F	-	+5.8	27

Nylon textiles are coated with a silicone oil finish which is considered to be an important part of the textile. The silicone oil provides lubrication between the fibers and between the yarns in the textile and tends to counteract any tendency for layers of textiles to stick when the normal surface moisture is removed from the pressure packed landing system in a vacuum.

MIL-C-7020 specifies a silicone oil finish for nylon parachute cloth and states Dow Corning silicone emulsion ET-112A has proved satisfactory. (In textile processing the emulsion is broken down and silicone oil remains.)

The following conclusions were made in a Northrop Ventura analysis of ET-112A silicone oil finish (Ref. 28):

1. The oil is a linear dimethyl polysiloxane

2. A vapor pressure of 10^{-12} torr at room temperature (quoted by Dow Corning) represents the vapor pressure of the lighter fractions of the oil.

3. Assuming a vapor pressure of 10^{-12} torr for the silicone oil and a temperature of 120°F, a fabric exposed directly to a vacuum of infinite size (or a pump with infinite capacity) would lose all its silicone oil in 5.3 days.

4. Assuming a vapor pressure of 10^{-12} torr for the silicone oil, a temperature of 120°F and a vacuum outside the Apollo landing system compartment, a main parachute within the compartment would lose all its silicone oil in 1.5×10^{7} days.

It is reasonable to assume that a substantial portion of the finish provided by ET-112A will remain on the textiles at the conclusion of the ELOR mission. However, MIL-C-7020 does not require that the silicone oil finish be compatible with a vacuum environment. It is possible that a relatively high vapor pressure silicone finish could be used to provide the finish, and that it would be substantially lost during ELOR mission. Finish requirements should be further investigated and better defined and it should be determined what, if any, corrective action is necessary to assure the integrity of the textile finish during the ELOR mission.

Pyrotechnics

The pyrotechnics in the present Apollo ELS are sealed to a specified maximum leak rate of 10^{-8} standard cc/sec of krypton. This is equivalent to 2.7×10^{-8} standard cc/sec of air. In a vacuum, air and water vapor will be lost from the pyrotechnic units at a slow rate. A reduced pressure within the units will tend to slow the operation of the units, which is no disadvantage. Delay times and rise times will tend to increase, although changes will be quite small. As pressure within the cartridges decreases, the more volatile components within the units will volitilize. However, the low leak rate will limit the loss of material to an infinitesimal quantity in the time period of the ELOR mission (Ref. 5).

In a Northrop Ventura analysis of the vacuum stability of the reefing line cutter (Ref. 29) it was concluded that the cutters contain materials which cannot withstand direct exposure to a high vacuum for an extended period of time. For example, nitroglycerin (in Hercules "Unique") has a vapor pressure of 7.3×10^{-3} torr at 122°F and 2.5×10^{-4} torr at 68°F. Potassium chlorate (in the M-42 primer) has a calculated decomposition pressure of 6.1×10^{-3} torr of oxygen at 81°F (calculated from standard free energies).* The rate of decomposition was not included in the analysis. It may be very slow.

The leak rate of each individual pyrotechnic unit is tested, and it can be concluded that the present pyrotechnics are suitable for the Extended Lunar Orbital Mission. However, it is probable that the development of gross leakage would effect the functional characteristics of a unit.

Effects of Penetrating Radiation Environment

It was noted in Section 2.3.4 that a radiation exposure of 500 rads can be expected under a shield of 1 gram per square centimeter of aluminum from two passages through the earth's radiation belts. In addition, there is a probability of 0.0070 of exposure to one or more solar flare events producing an exposure of 7×10^{-3} rads under the same shielding during a 99 day mission. One gm/cm^2 of aluminum is the nominal radiation shielding of the ELS compartment (Ref. 5).

Teflon is the most radiation sensitive material within the ELS compartment. Threshold damage in air is produced by an exposure of about 10^{-4} rads (Ref. 5). An exposure at least an order of magnitude higher is required to produce significant damage in a vacuum. An exposure greater than 10^{-5} rads would be required to produce threshold damage to the other materials in the compartment.

Threshold damage to the most radiation sensitive compartments in the sequence controller is about 10^{-4} rads (Ref. 5). However, the sequence controller is located in the crew compartment, with a nominal shielding of 5 gms/cm^2 of aluminum. This means the sequence controller will be exposed to one to two orders of magnitude less radiation than components in the landing system compartment (Ref. 5).

It can be concluded that the ELS can withstand more than 10 of the quoted solar flare events without sustaining significant functional degradation. The ELS is far more tolerant to penetrating radiation than the crew, and it is assumed that any manned mission would be aborted long before the improbably accumulation of 10 major solar flare events.

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*A decomposition pressure of 6.1×10^{-3} torr means that if the partial pressure of oxygen falls below 6.1×10^{-3} torr potassium chlorate will decompose to produce oxygen until the equilibrium pressure of oxygen (6.1×10^{-3} torr) is restored.

Effects of Meteoroid Environment

There is a probability of 0.9997 that the ELS compartment wall will not be penetrated by a meteoroid during the ELOR mission.

Penetration does not necessarily mean the ELS will be damaged. Even if damaged, it does not necessarily mean the ELS will not function. Even extensive damage to one main parachute could leave the other two main parachutes and other essential components functional. However, there is a small, but finite, probability the meteoroid damage will degrade the landing system to an extent where it will not function.

Effect of Prolonged Parachute Packing Stresses

Apollo ELS parachutes are pressure packed and are required to be usable without repack for a period of one year (Ref. 6).

The pressure packed parachutes are under considerable compressive stress and the retention assembly is under tensile stress during the period of storage. The U.S. Army, Navy, and Air Force, have tested the storage life of several types of parachutes (some pressure packed) under various earth ambient conditions. The results indicate that pressure packed parachutes can be stored for several years under earth ambient conditions (Ref. 5).

Storage during a mission involves the vacuum, radiation and temperature environment inside the ELS compartment. Very little is known about the effects of this environment on the life of textile materials under compressive and tensile forces. However, the present Apollo ELS is considered suitable for storage in a pressure packed condition for a 14-day mission. On this basis, an extension of mission time to 99 days appears reasonable (Ref. 5).

Sequence Controller

The sequence controller is located in the crew compartment. Effects of this environment on the sequence controller are expected to be insignificant over the duration of the ELOR mission.

The metallic diaphragm of the baroswitch is exposed directly to ambient space conditions. The only effect to be expected from this exposure is an insignificant loss of metal by evaporation.

RELIABILITY ANALYSIS

The Apollo Earth Landing Systems, Increased Capability, was used as a basis for the reliability analysis of the Apollo Extended Mission. The reliability math model which was applicable to the analysis is contained in NV Report, NVR-6249, Apollo Earth Landing System, Increased Capability, Report of Reliability Analysis. The model detailed to the major component of functional level is shown in Figure W-1. The results of the analysis are presented in Section 4.1.9 of Volume I of this report.

In the course of the reliability analysis, the following rules were applied.

1. The criterion for successful Parachute Subsystem operation was as follows: descent on two of the three inflated main parachutes met the maximum rate of descent requirements of 35.0 fps at sea level.

2. All data from laboratory and field development (design verification), qualification, acceptance tests and WSMR and KSC launches were used in the reliability assessment, where fit, form and function of the test specimen were compatible with the final qualified configuration.

3. Failures followed by successful corrective action were not assessed in the system model while previous successes were assessed, if it could be concluded that the remaining equipment operating parameters were not affected by the corrective action.

4. Only primary functional failures were recorded in the system model. Secondary failures, such as specification requirements that could not be directly related to system failures, were omitted.

5. The reliability of each basic component was estimated with a 50 percent confidence level for calculation of the operational reliability.

6. The reliability value of 1.0 was taken for contractor-furnished equipment and astronaut manual-input functions.

7. On items of which reliability varied with applied loads and for which less than 5 data points were available which showed ultimate strength, a standard deviation of 5 percent from the mean value was assumed.

8. In items of which reliability varied with applied loads, modifying design factors such as material strength degradation due to temperature, stitching, abrasion, etc., were applied to derate analytically derived component strength or mean demonstrated strength whenever such degradation factors had been excluded from testing of an item.

Analysis of Potential Failures

The Apollo ELS is presently qualified for a 14 day mission, and only those potential failure modes which are related to extended lunar orbital storage will be considered in this section.

System failure because of faulty pyrotechnic function will be considered first (Figure W-2). Causes of failure could be (a) failure of the sequence controller, (b) vacuum degradation of the pyrotechnic units, or (c) degradation of Viton seals.

Failure of the sequence controller is considered improbable because it is exposed only to the crew compartment environment and both parallel and series redundancy are provided in the controller to assure proper function and to prevent early firing.

Vacuum degradation of the pyrotechnic units would require the development of gross leakage in the units during the mission. If this occurred in a reefing cutter the effects would be minimized because of the use of redundant reefing cutters. However, equivalent redundancy is not employed with the pressure cartridges. Each pyrotechnic unit is tested at the time of manufacture for compliance with the specified maximum leak rate, and it is assumed that present pyrotechnic design and acceptance procedures are adequate for the current 14-day Apollo mission. Development of increased leakage during the extended mission is not anticipated. However, in the absence of direct evidence that degradation would not occur. one or more of the following compensating actions are recommended:

1. Test the pyrotechnic units to verify in-specification function after exposure to a 99 day thermal/vacuum environment.

2. Conduct further analysis and tests to determine the vacuum stability of the seals and the pyrotechnic components.

3. Investigate methods to provide redundant sealing of the pyrotechnic units.

Action (1) would provide direct evidence of suitability. An increased understanding of any potential problems would be provided by action (2). Action (3) is relatively undesirable in that it would leak to at least a minor redesign of the pyrotechnics.

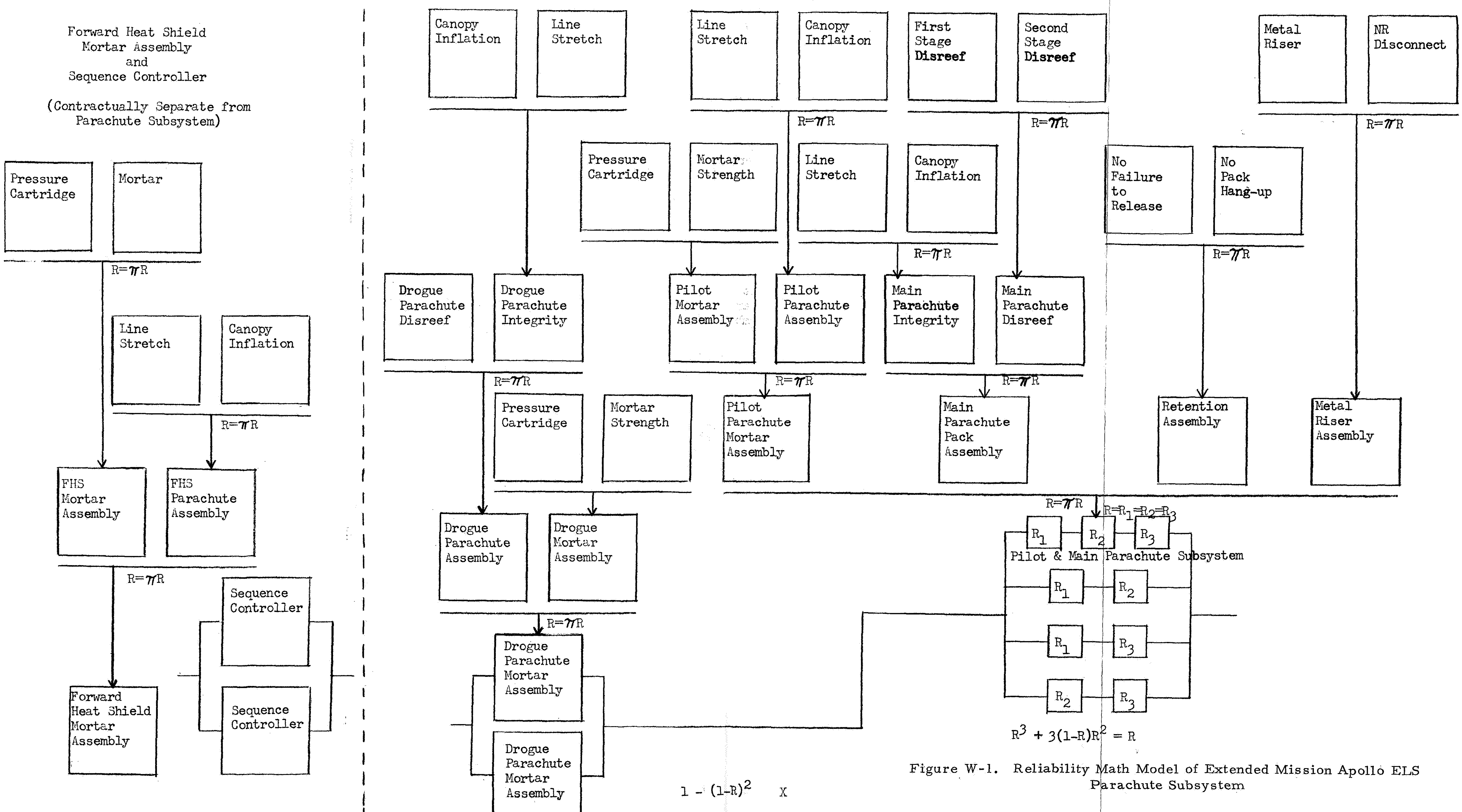


Figure W-1. Reliability Math Model of Extended Mission Apollo ELS Parachute Subsystem

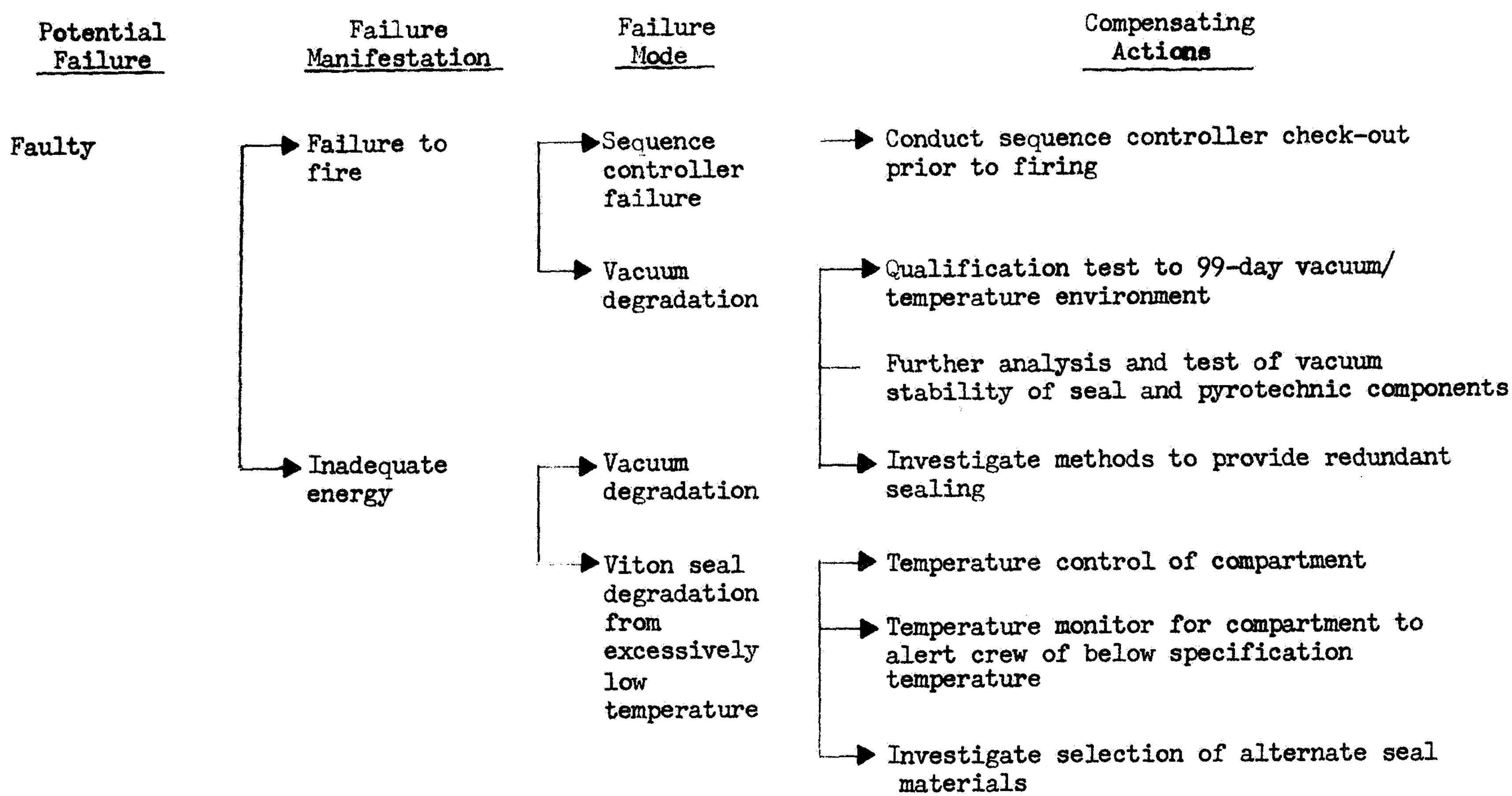


Figure W-2. Potential Pyrotechnic Failure

Viton seal degradation is a possibility if compartment temperature falls below the minimum specification temperature. Alternate compensating actions are:

1. Temperature control of compartment
2. Temperature monitor to alert crew if the lunar storage mode is producing below specification temperatures in the compartment.
3. Investigate the feasibility of alternate seal materials.

A combination of actions 1 and 2 is recommended. Action 3 is less desirable because elastomeric seal materials, in general, become brittle at low temperatures. Also, action 3 could reasonably lead to a requirement for modification of the present design.

An analysis of potential landing system failure because of structural textile failure or steel riser cable failure is summarized in Figure W-3. Again, the only potential failure modes which are considered are those which can be directly related with extended lunar storage. Parachute failure can occur without a system failure because parachutes are redundant, except for the forward heat shield parachutes (and failure of these parachutes does not necessarily lead to system failure.)

A reduction in textile strength could occur if the compartment temperature remained consistently well above the maximum specification temperature during lunar storage. Alternate compensating actions are:

1. Provide adequate temperature control for the compartment to assure temperatures will not substantially exceed the maximum specification temperature.
2. Provide a temperature monitor for the compartment to alert the crew on the lunar surface if compartment temperatures are exceeding maximum specification temperature.
3. Upgrade textile strength to compensate for any irreversible thermal degradation during lunar storage.

Actions 1 and 2 are recommended.

A reduction in textile strength could result from vacuum degradation. Alternate compensating actions are:

1. Further analyze and test to better define and verify the stability of textiles (nylon in particular) to extended exposure to a thermal/vacuum environment.

Potential Failure	Failure Manifestation	Failure Mode	Compensating Actions
Parachute Failure	Textile failure on deployment	Reduction in textile strength from excessive storage temperatures	Temperature control of compartment
			Temperature monitor for compartment to alert crew of consistent above specification temperature.
			Up-grade textile strengths
		Reduction in textile strength from vacuum degradation	Further analysis and test of stability of nylon to vacuum/ temperature environment
			Up-grade textile strengths
		Reduction in textile strength from loss of finish	Analysis of finish requirement and up-grade textile specifications
			Reduce compartment venting to minimum feasible extent
			Add excess silicone to compartment
	Steel riser cable failure	Cold welding in vacuum	Verify that present processing provides film of dry lubricant on all cable strands.

Figure W-3. Potential Parachute Failure

2. Upgrade textile strengths to compensate for possible strength reduction.

Action 1 is recommended. Action 2 is less desirable because significant vacuum strength degradation has not been demonstrated, required system reliability is exceeded even if there were a 10 percent loss in textile strength and a redesign of the present landing system is undesirable.

Loss of the silicone finish on nylon textiles could degrade the textiles because of increased frictional damage when a load is applied and because adhesion of nearly moisture-free pressure packed textiles would be encouraged. Alternate compensating actions are:

1. Analyze nylon finish requirements and upgrade specifications to assure the use of silicone oils with the lowest practical vapor pressure.

2. Reduce compartment venting to the minimum practical level.

3. Add excess silicone oils to the compartment to maintain a silicone oil partial pressure in the compartment.

A combination of actions 1 and 2 is recommended. Action 3 is less desirable because of the weight penalty.

Cold welding of the steel riser cable strands would inhibit motion of the strands when a load is applied. This could degrade the strength of the cables. The compensating action is testing to verify that current production methods do provide a dry lubricant coating to all cable strands – in particular the internal strands of the cables.

APPENDIX X

EMERGENCY CM FUEL CELL COOLING THROUGH WATER SUBLIMATOR*

Should it be necessary to supplement the cooling capacity of the Block II radiator, a water sublimator can be added, as indicated in Figure X-1, for temperature control at elevated power. The sublimator is located in the coolant circuit prior to the regenerator rather than the condenser for simplicity of control. It uses water supplied through a temperature controlled valve from the ECS storage. Unmanned operation in conjunction with the power profile provides a water supply in excess of the sublimator requirements, as shown in Figure X-2. Although the power demands are not yet accurately defined it appears the sublimator would need only about 43 percent of the water produced during the unmanned period. The purity of the water from P&WA low temperature fuel cells is comparable to that obtained from the Apollo PC3A-2. Water from either type of fuel cell is considerably more pure than ordinary tap water.

The operation of the sublimator is determined by radiator return temperature level. Under normal base load conditions, proper coolant temperature can be maintained by the space radiator. Under the worst conditions of environment and power output, our calculations predict an excessive radiator exit temperature level. When the coolant temperature reaches the limiting value, a controlled water flow is released to the sublimator and a bypass valve regulates the temperature level at the inlet to the powerplant regenerator. A decrease in radiator exit temperature below the sublimator operating threshold results in a termination of the water flow. The thermal capacity of the radiator exit coolant flow completes the sublimation of any residual water and prevents freezing of the unit.

A water boiler can also be considered instead of a sublimator to augment the radiator, but the level of development on the sublimator has progressed to the state where it would appear to be less complex in control and more adaptable to this application. Sublimators are currently being used in the Apollo program.

*Provided by Pratt and Whitney Corp. through Reference 4.16 of Vol. I.

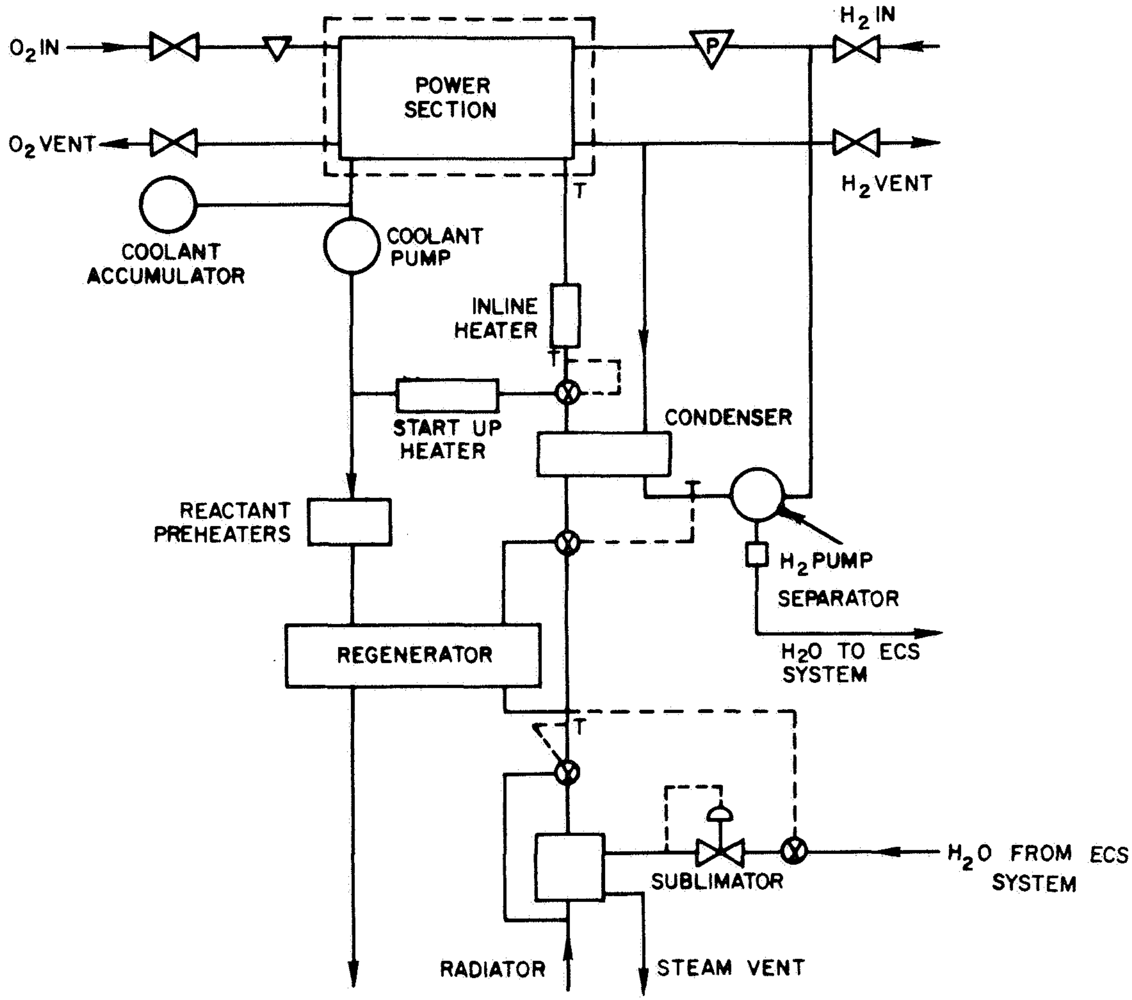


Figure X-1. Powerplant System Schematic for CSM ELOM Application, Sublimator Added to Supplement Radiation

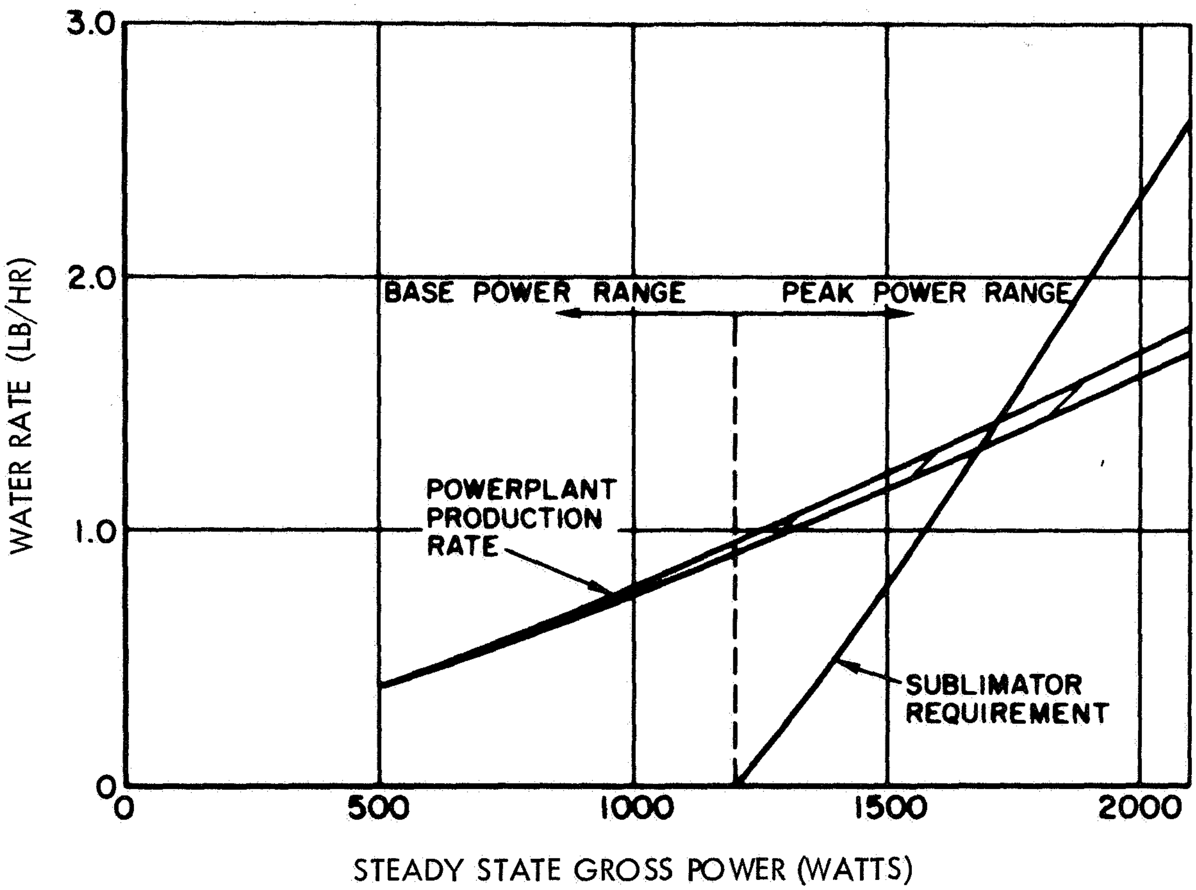


Figure X-2. Sublimator Requirements for CSM ELOM Application

APPENDIX Y

IMPROVED CM COMMAND DEMODULATOR DECODER AND LUNAR SURFACE TO CSM COMMAND LINK*

The Command Demodulator Decoder (CSS) referenced in Section 4. 1 of Volume I and described in Section 4 of Appendix I was found to be a mission and crew critical function. Loss of the function could be catastrophic to the orbiting CSM and subsequently the crew.

The analysis of its contribution to P_{90} and P_s indicated potential weakness that could only be corrected through design. For that reason Collins Radio was requested to take a second look at the CDD and the related remote control function. The results are reflected in the following.

The CDD as described in Appendix I will be modified as reflected in Figure Y-1. The revised configuration employs dual active redundancy for the demodulator and a portion of the decoder.

The word decoding logic as shown back in Figure I-4 will have two additional decoded command outputs which will be used to control selection of either one of the two memory/relay driver sections of the decoders. The memory and relay driver sections of the decoders cannot be used in a dual active redundancy configuration since a failure in the output of either of the two, the memory or relay driver, would result in a failure of the associated output command channel. Active redundancy of these elements could only be employed in a majority vote configuration which is not considered practical due to equipment complexity. The reliability of the OR gate section of the unit is increased by activation only during periods when a command message is being decoded.

Operationally, the command link would function as follows:

1. The astronaut in the LM would initiate a command which would be transmitted to the CSM.

2. The CSM receives, demodulates and decodes the command signal providing an output control to the designated relay(s). The verification circuit routes a signal to the SME data link which transmits the verification data to the LM.

*Provided by Collins Radio Co.

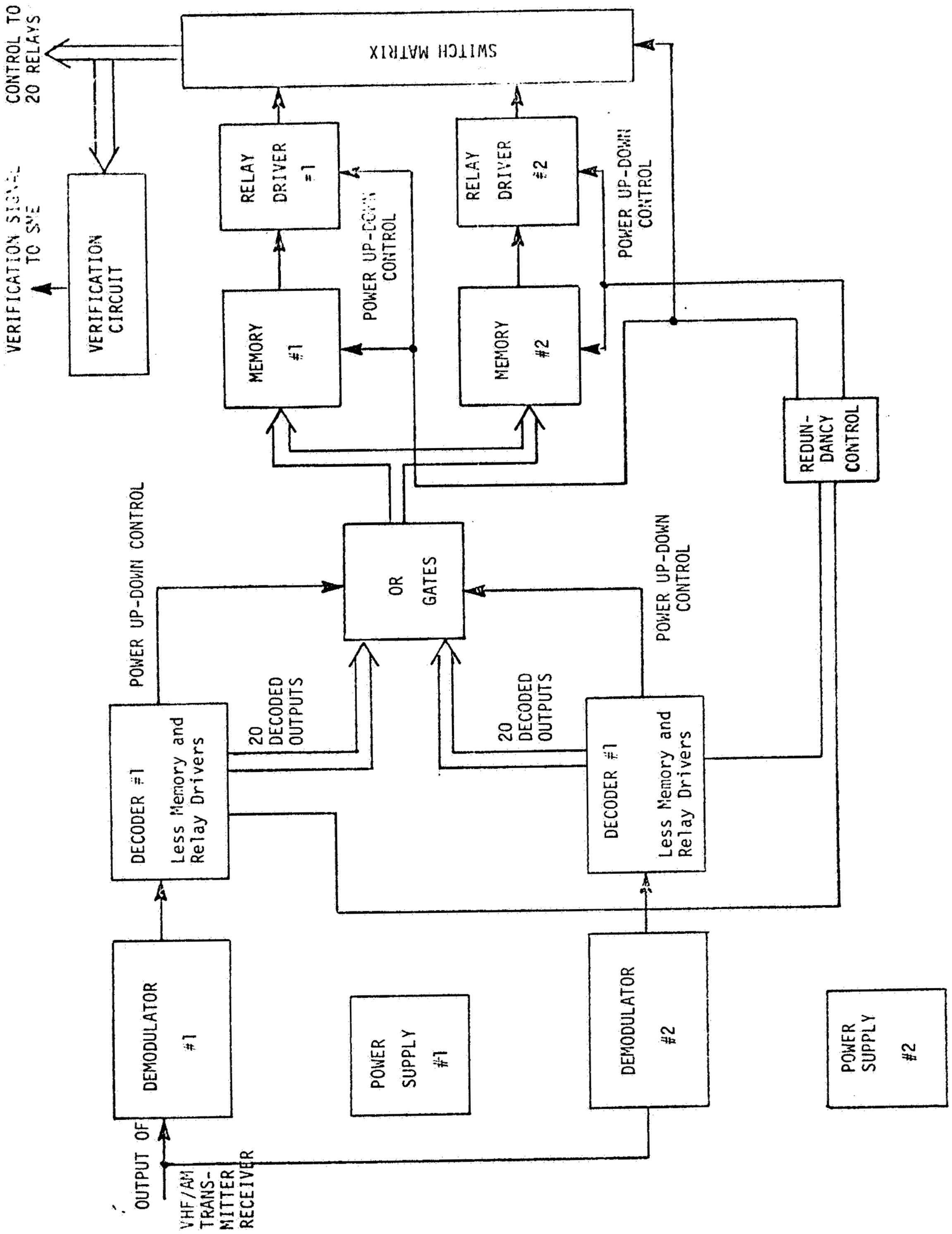


Figure Y-1. Block Diagram of Redundant Command Demodulator-Decoder

3. The astronaut in the LM monitors the SME data to verify receipt and execution of the command signal by the CSM.

4. If the verification message is received by the LM, the astronaut is virtually assured that the command had been executed in the CSM. If the verification message is not received, then either an error was introduced during transmission or a failure had occurred in the memory, relay driver, switch matrix, verification circuitry or the SME data link assuming that only a single failure had occurred. An SME data link failure will very likely be evident by examination of other SME data channels. If a failure in the SME had occurred, the astronaut could switch to a standby redundant SME as described in our study report. To eliminate the possibility that a transmission error was the problem, the astronaut transmits the desired message again. If the verification message resulting from the second command transmission is not received, then the astronaut would switch to the alternate memory/relay driver by an up command.

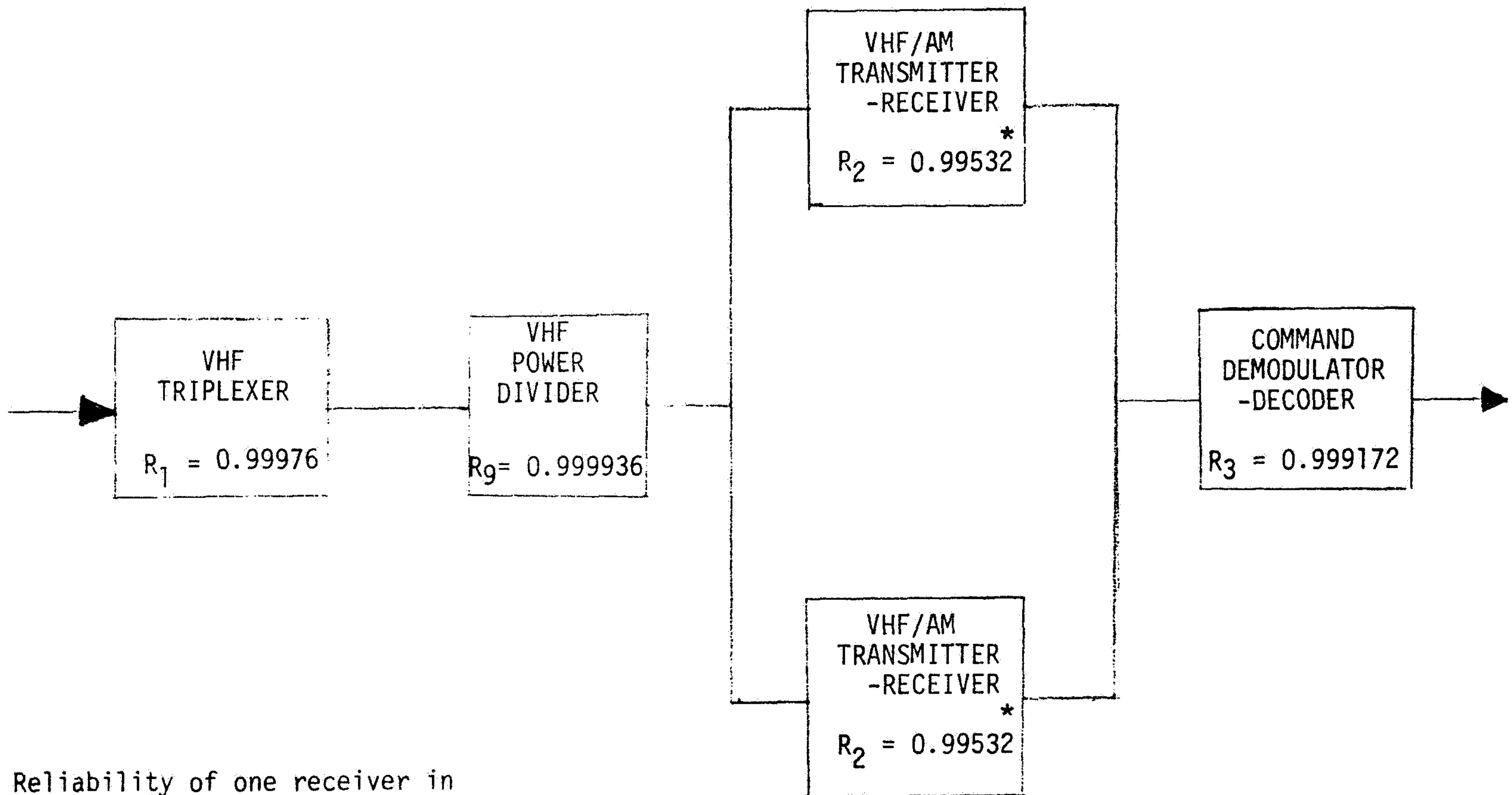
5. The astronaut then retransmits the desired command. If the verification signal is still not received by the LM, then either the verification circuit had failed or a multiple failure in the demodulator section or dual active redundant portion of the decoder had occurred. Since the verification circuit is non-redundant, no remedial action could be taken. However, commands can still be transmitted to the CSM and perhaps verification can be made by investigating the changes in the SME data channels. If the verification signal is received, then the astronaut knows that the command signal had been properly executed.

It must be pointed out that when switched from one memory/relay driver section to the other the state information of all relays has been destroyed and must be reset to their desired states.

The reliability of the Command Demodulator-Decoder, with the internal redundancy, is 0.999172. This yields an overall subsystem reliability for step five of 0.997553. The reliability of the command function is 0.994257 and is now limited by the reliability of the VHF/AM Transmitter-Receiver.

The improved Command Demodulator-Decoder's weight will be nine pounds and will have a volume of 300 cubic inches. The power requirements will be 7 watts during operation and 5.5 watts during standby.

A second VHF/AM Transmitter-Receiver can be added to the subsystem to further improve the reliability of the lunar surface/LM to CSM command link. The reliability logic of this configuration is shown in Figure Y-2.



* Reliability of one receiver in the VHF/AM Transmitter-Receiver Equipment.

Figure Y-2. Reliability Logic Diagram of Improved Lunar Surface LM-to-CSM Command Link

A power divider is required to split the single RF input to the two receivers. The outputs of the receivers are combined in the Command Demodulator-Decoder.

The performance of the lunar surface/LM to CSM command link will be degraded slightly since the input signal will be 3 db poorer into each receiver. The non-linear characteristics for the AM detector for low input carrier-to-noise ratio results in approximately a 1.5 db loss in circuit quality at the maximum operating range after combining in the Command Demodulator-Decoder. Since the circuit margin for the lunar surface/LM to CSM command link is 8.6 db (nominal), this degradation will not affect the required performance of the link. A possibility also exists that the circuit quality may be degraded even further for certain failure modes of either of the two receivers. A loss of up to 7.5 db can be encountered if one receiver fails and provides a noisy output but the circuit margin remains sufficient to handle this situation.

Figure Y-2 shows the reliability logic for the Lunar Surface/LM to CSM Command Link incorporating reliability improvement steps 5 and 6. We estimate that the reliability of the VHF power divider to be 0.99994 and the parallel combination of the receivers to be 0.99998. The overall subsystem reliability with the redundant VHF equipment and improved Command Demodulator-Decoder is 0.998. The reliability of the command function alone is 0.9991.

The additional VHF/AM Transmitter-Receiver will weigh 13.5 pounds and have dimensions of 4.7" x 6" x 12". The input power required by the additional VHF unit will be 2 watts for the receive-mode of operation.

The VHF power divider is estimated to weigh 8 ounces and will have a volume of 10 cubic inches.

APPENDIX Z

ADDITIONAL DATA, P&W FUEL CELLS FOR THE ELOR CSM*

The P&W PC8 Electrical Power Plants are proposed for use in the ELOR CSM. Details are presented in Section 4.2.1 of Volume I. The following is presented as additional data.

The weight of each of the three powerplants is 130 pounds nominal. If it is desired to attach the powerplant to the vehicle in the same manner as the present Apollo powerplant, a vehicle mount cone weighing 21 pounds is available. If a sublimator is used to supplement the Block II radiator, its weight is approximately 8 pounds.

Based on presently available low temperature matrix cell performance, the estimated voltage characteristic at the end of mission is indicated in Figure Z-1 for a powerplant incorporating a 32 cell powersection mated to PC3A-2 control components. The corresponding maximum hydrogen consumption rate for the powerplant is indicated in Figure Z-2. Oxygen consumption is 7.94 times the hydrogen consumption. Purge flows are additional, but they amount to less than 1 percent of the total reactant supply.

The performance used in this study is a conservative assessment of the performance data obtained from tests on P&WA low temperature cells. If reactant consumption becomes critical, better performing cells are available. Specific reactant consumption is inversely proportional to cell voltage at a given current density.

Powerplants using PC8 cells have shown their ability to tolerate prolonged short circuits (Figure Z-3) and repeated spike loads (Figure Z-4). They have also been operated by NASA in parallel with Apollo batteries (Figure Z-5).

*Provided by Pratt and Whitney Co.

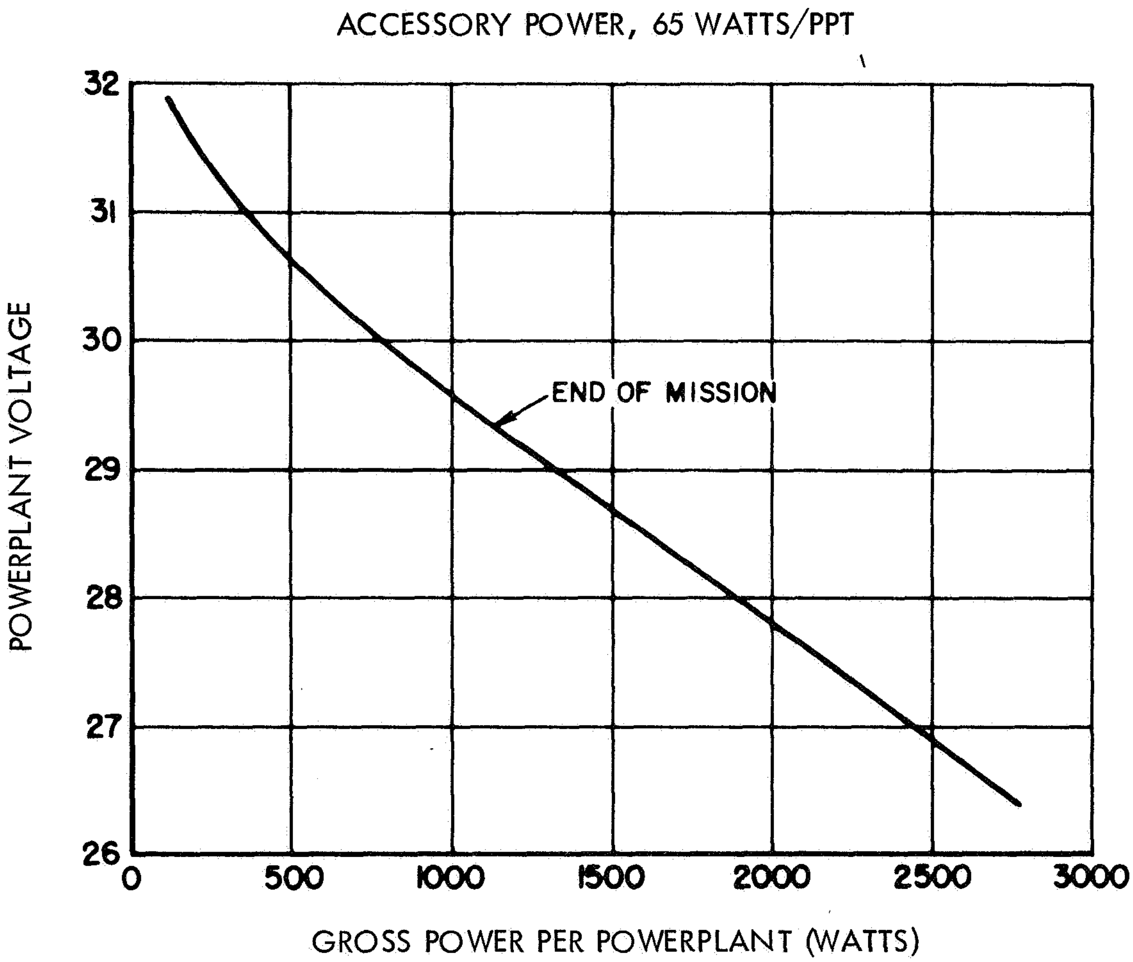


Figure Z-1. Voltage Characteristic for CSM ELOM Application

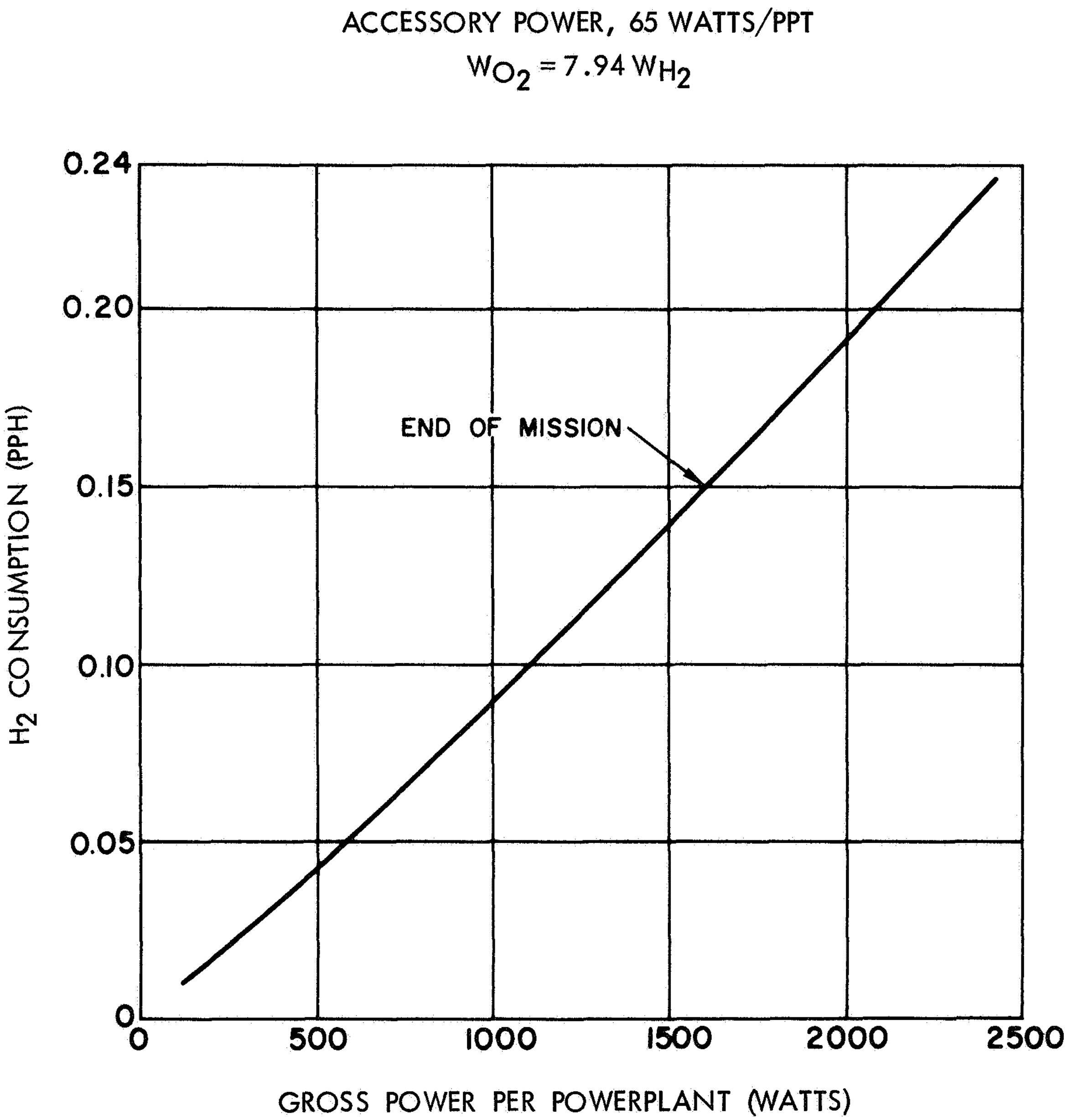


Figure Z-2. Hydrogen Consumption for CSM ELOM Application

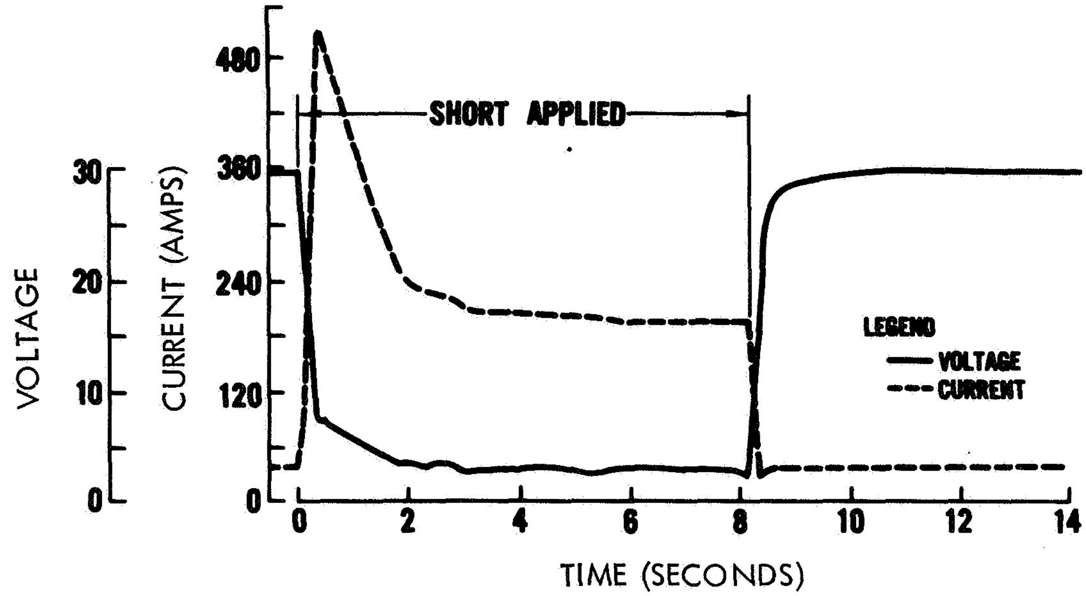


Figure Z-3. PC8A-2 Short-Circuit Tests

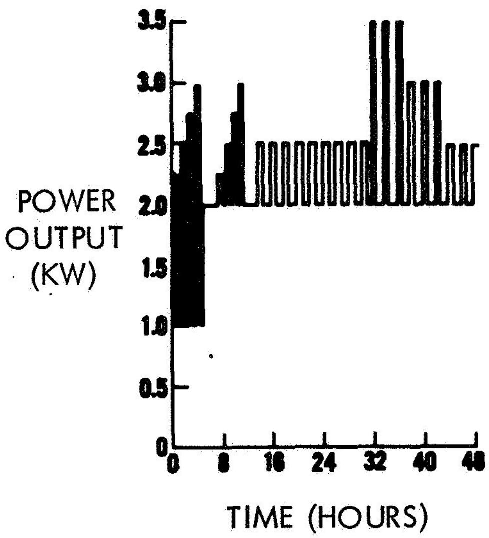


Figure Z-4. Fuel-Cell Spike Load Test

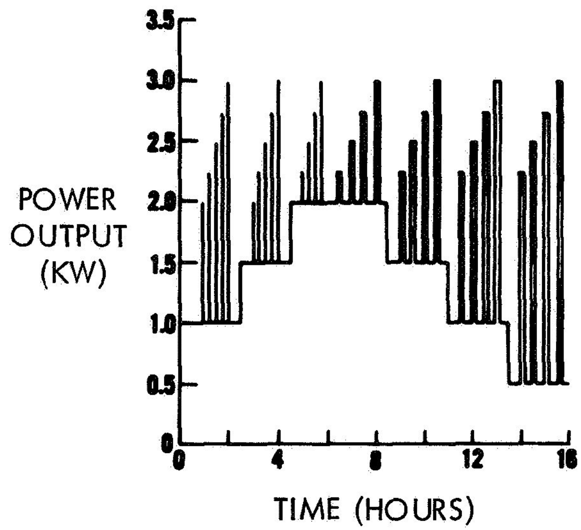


Figure Z-5. Fuel-Cell and Battery Parallel Operation